19890017534

591292

97p.

NASA Contract Report 182280
RI/RD88-256

Space Station Hydrogen/Oxygen Thruster Technology

G.L. Briley and R.S. Iacabucci

Rockwell International
Rocketdyne Division
Canoga Park, California

November 1988

Prepared for
Lewis Research Center
Under Contract NAS3-25142

National Aeronautics and
Space Administration

FOREWARD

This document contains a detailed summary of all tasks performed under this contract. Included are design description, design analysis, fabrication procedures, hot-fire test data and analysis, and conclusions. This document is submitted in fulfillment of the Final Report Data Requirement of Task VI of Contract NAS 3-25142.

ABSTRACT

This report covers the effort expended by the Rocketdyne Division of Rockwell International in fulfilling the requirements of the Space Station Freedom Hydrogen/Oxygen Thruster Technology program. The report includes the basis and the rationale for the design of the thruster, injector, and nozzle; discusses the test and results; and presents the lessons learned, together with conclusions and recommendations for the development of the Space Station Freedom thrusters.

CONTENTS

FIGURES

FIGURES (Continued)

FIGURES (Continued)

TABLES

1.0 INTRODUCTION AND SUMMARY

The primary propulsion requirements for the manned space station are long life, reliability, and low maintenance as dictated by safety and life-cycle cost considerations. The Space Station Freedom Phase B studies by the National Aeronautics and Space Administration (NASA), Rocketdyne, and the Phase B contractors indicated that gaseous oxygen/gaseous hydrogen (GO_2/GH_2) supplied by electrolysis of water would offer significant advantages for the Freedom Station, when compared to other candidate propulsion systems. The hazard and contamination levels of GO_2/GH_2 are inherently low by comparison with monopropellants or storable bipropellants, and the compatibility and ease of integration with other systems of the Freedom Station provide a high degree of synergism. The integration of the GO_2/GH_2 propulsion system into the Freedom Station systems and supply logistics program, including off-loading orbiter excess water, will eliminate the need for supplying propellant to the space station. The GO_2/GH_2 system is the lowest life-cycle cost, by significant margins, of all systems studied.

As an outgrowth of the Freedom Station Phase B studies and results of Rocketdyne company-funded effort, which was initiated in 1984 for low-thrust GO_2/GH_2 rocket engines, Rocketdyne was awarded a contract by NASA-Lewis Research Center (LeRC) in March 1987 to design, fabricate, and deliver for evaluation a GO_2/GH_2 thruster.

The program consisted of two phases comprising the following tasks:

- Phase I: Preliminary and Final Design, Fabrication, and Testing
    - Task I — GO_2/GH_2 Thruster Preliminary Design
    - Task II — Thruster Final Design
    - Task III — Thruster Fabrication
    - Task IV — Performance Optimization and Characterization
    - Task V — Delivery
    - Task VI — Reports

- Phase II (Option): Fabricate and Test Second Thruster
    - Task VII (Option) Fabricate Second Thruster
    - Task VIII (Option) Performance Optimization and Characterization
    - Task IX (Option) Long-Life Testing
    - Task X Delivery

The contract start date was 5 March 1987, with funding allocated for Phase I. The contract was amended on 30 April 1987 to perform Tasks VII through X of the Phase II Option.

The major program milestones and their completion dates are listed in Table 1-1.

Table 1-1. Program Milestone Completion Dates

Task	Milestone	Completion Date
I	GO_2/GH_2 thruster preliminary design	25 March 87
II	Final design review	25 March 87
III	Complete fabrication and assembly	20 August 87
IV	Deliver thruster to NASA-MSFC for test Complete characterization tests	28 August 87 3 March 88
V	Deliver thruster to NASA-LeRC	28 September 88
VI	Deliver final report to NASA-LeRC	15 November 88
VII (Option)	Complete fabrication of second thruster	19 March 88
VIII (Option)	Complete characterization tests	10 March 88
IX (Option)	Complete long-life tests	(at LeRC Facility)
X	Deliver second thruster to NASA-LeRC	6 October 88

The basis of the thruster design for this program was the configuration emanating from the Rocketdyne GO_2/GH_2 prototype thruster program initiated in 1984. The program successfully demonstrated 87,399 s (24.3 h) of firing time over a mixture ratio range from 3.1 to 8.1 and 10,500 thrust pulses of approximately 0.5-lb•s impulse each. The design was ready for preliminary and final design reviews at program start.

Fabrication was performed at Rocketdyne. Vendors were used for selected detail part machining. Checkout, calibration, cold flow, and assembly operations were performed in the Rocketdyne Engineering and Material Laboratories. Thruster hot-fire testing was performed at Marshall Space Flight Center (MSFC), Huntsville, Alabama, in the test stand 302 vacuum chamber and test facilities. The Oxygen/Hydrogen Propulsion Systems Test Bed, Contract NAS8-36418 (Reference 1), was installed in the facility during the performance of this contract effort. The 302 facility and the test bed were used to perform the thruster hot-fire testing.

One hundred and four tests were conducted to provide data for performance optimization and characterization of the two thrusters produced. Included were several tests conducted with the original Rocketdyne prototype thruster hardware and an existing "low-heat flux" injector, designed and fabricated by Rocketdyne. These latter tests anchored the data from the new units to previous work and provided information to assist in the production of the flight thruster design, performance, and life.

Table 1-2 summarizes the hardware configurations and test experience to date with the Rocketdyne 25 lbf hardware. Four injectors and three nozzles have been tested for a total of 216 steady-state tests and 10,451 thrust pulses. Testing time of 25.6 hours has been accumulated over a propellant mixture ratio range of 3.1 to 8.5. The life, performance, and pulsing capability of the thruster has been demonstrated. Hardware characteristics and configurations to enhance durability and life without performance degradation have been defined.

Table 1-2. 25 lbf Thruster Test Summary and Background Test Experience

INJECTOR	NOZZLE	# OF TESTS	DURATION (sec)	Pc (psia)	MR	RESULTS
PROTOTYPE	PROTOTYPE	121 10,451 PULSES	87399	45 - 106.8	3.1 - 8.1	LIFE/PERFORMANCE/PULSING DEMONSTRATION
PROTOTYPE	LeRC 1	22	135	99.3 - 114.4	6.0 - 8.0	PERFORMANCE VERIFICATION WITH NEW NOZZLE
PROTOTYPE	LeRC 2	4	155	103.7 - 111.2	6.0 - 8.1	PERFORMANCE VERIFICATION WITH NEW NOZZLE
LeRC 1	LeRC 1	26	1866	52.0 - 147.0	3.1 - 8.3	PERFORMANCE/OPERATION VERIFIED ON NEW ASSEMBLY
LeRC 2	LeRC 2	20	1324	48.6 - 142.3	3.2 - 8.4	PERFORMANCE/OPERATION VERIFIED ON NEW ASSEMBLY
LHF	LeRC 1	23	1376	46.8 - 136.0	3.2 - 8.5	Is vs %BLC ESTABLISHED LOW SKIN TEMP/LONG LIFE PRE
4 INJECTORS	3 NOZZLES	216 10,451 PULSES	92142 (25.6 HRS)	45.0 - 147.0	3.1 - 8.5	CAPABILITY DEMON

2.0 REQUIREMENTS

The design and performance requirements for the GO_2/GH_2 thruster established by the contract are presented in Table 2-1. A chamber pressure of 100 psi, an expansion ratio of 30:1, and thrust chamber regenerative cooling were chosen as the nominal design points.

Table 2-1. Summary of Thruster Design Parameters

Parameter	Requirement
Design mixture ratio	8.0
Mixture ratio range	3.0 to 8.0
Life capability	2×10^6 lb-s, minimum
Specific impulse (@ MR 8.0)	346 lbf-s/lbm
Minimum impulse bit	5 lbf-s/lbm
Propellant temperature	80°F, maximum
Thrust	25 lbf ±5lbf
Thrust throttle range	50% to 125% thrust
Chamber pressure	100 psia nominal
Nozzle expansion area ratio	30:1
Ignition and propellant valves	Integral flight type
Cooling technique	Regenerative cooling

3.0 THRUSTER DESIGN AND FABRICATION

This section summarizes the design and fabrication of the thruster and its component parts. The thruster, shown in cross section in Figure 3-1 and in external view in Figure 3-2, consists of a downpass regeneratively cooled thrust chamber; a coaxial injector assembly; individual fuel and oxidizer solenoid valves; and an igniter. (Appendix A contains detail assembly and component drawings.)

Minor modifications were made to the LeRC thruster components to incorporate lessons learned from the prototype efforts. These modifications are summarized in Table 3-1. The chamber hydrogen coolant channels were resized and increased in number from 24 to 30. The chamber hydrogen inlet manifold was simplified. The injector material was changed from 321 SS to 316 SS to enhance propellant compatibility. The injector oxidizer post recess was reduced from 0.080 in. to 0.060 in. to reduce the possible incipient erosion of the oxidizer post tips. The braze joint between the flange and combustion chamber was redesigned to improve integrity and fabrication. Chamber-to-injector details were changed to improve hot gas-sealing and chamber-to-injector centering characteristics. Details of the resulting combustor, injector, igniter, and valves designs are discussed in subsequent sections.

3.1 THRUST CHAMBER

The thrust chamber (Figure 3-3), consisting of a 1.5-in. long combustion chamber and a 2.81-in. long nozzle, is hydrogen cooled. The nozzle length is 80% of a 15-deg half-angle cone with an expansion area ratio of 30.

The thrust chamber is cooled by single downpass of hydrogen using the flow path shown in Figure 3-1. The incoming ambient hydrogen is introduced through dual inlets in the flange at the injector end of the thrust chamber and flows down the coolant passages to the nozzle end of the thrust chamber.

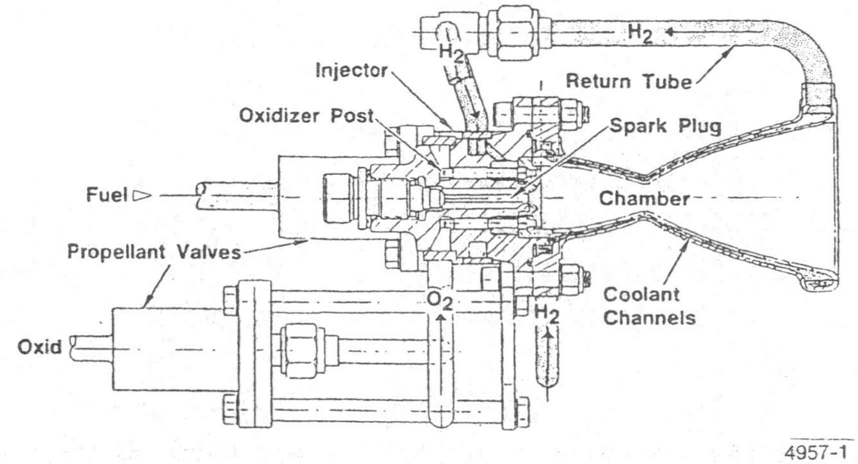


Figure 3-1. 25-lbf GO_2/GH_2 Thruster Assembly

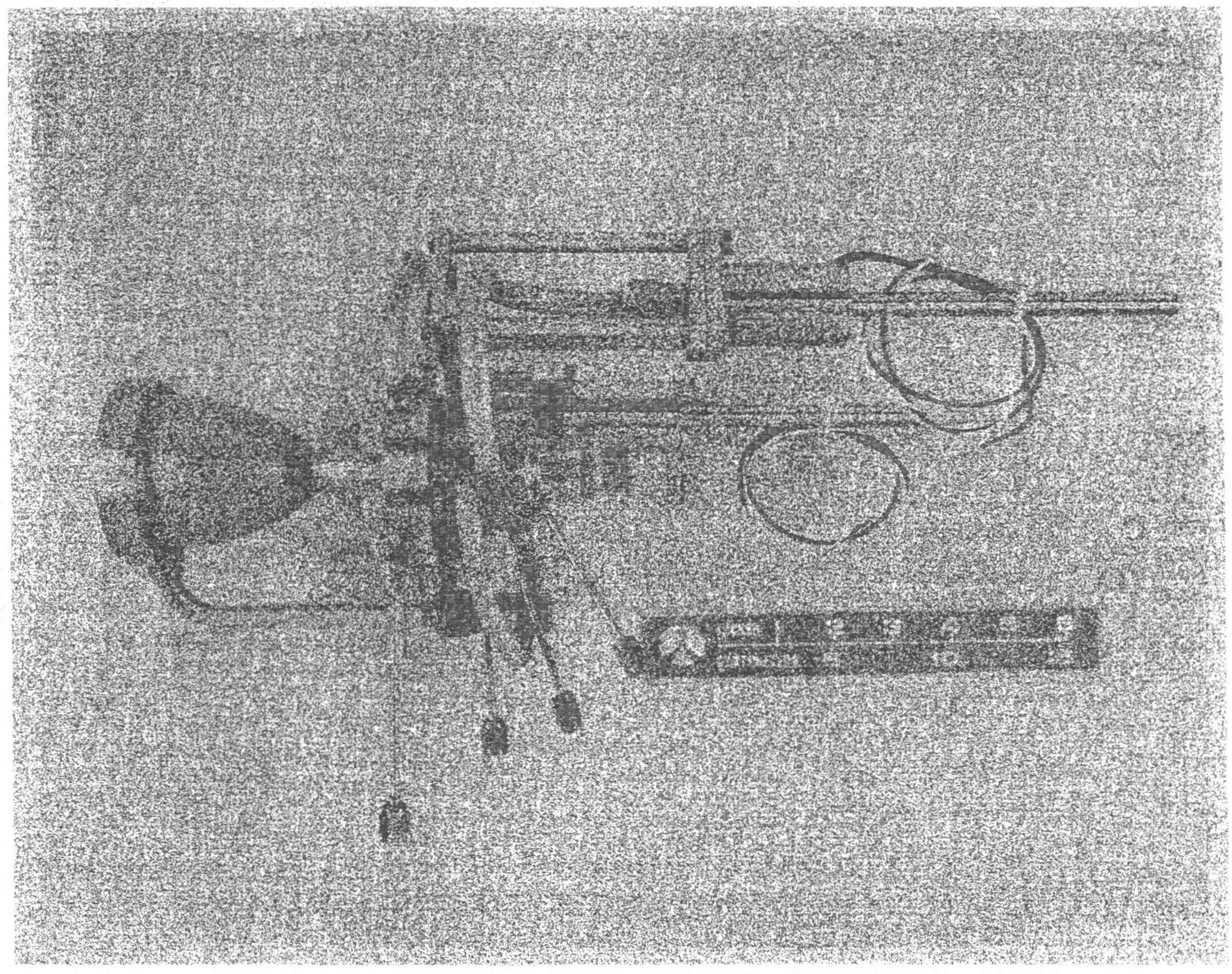


Figure 3-2. LeRC 25-lbf GO_2/GH_2 Thruster with Spark Plug
(15522-7/10/87-D18*)

Table 3-1. Summary of Modifications to Thruster from Prototype

Component	Prototype	LeRC Modification
Nozzle Coolant Channel		
• Number	24	30
• Width x depth	0.040 in. x 0.030 in.	0.020 in. x 0.030 in.
• Channel flow area (total)	(0.017 in^2)*	0.018 in^2
Nozzle Supply Manifold	Dual	Single
Injector		
• Material	321 SS	316 SS
• Oxygen post recess	0.080 in.	0.060 in.
Chamber to Injector		
• Seals	Dual seals	Single seal
• Centering	Bolts	Injector pilot OD

*With 0.025 in. wires inserted in channels to enhance heat transfer characteristics

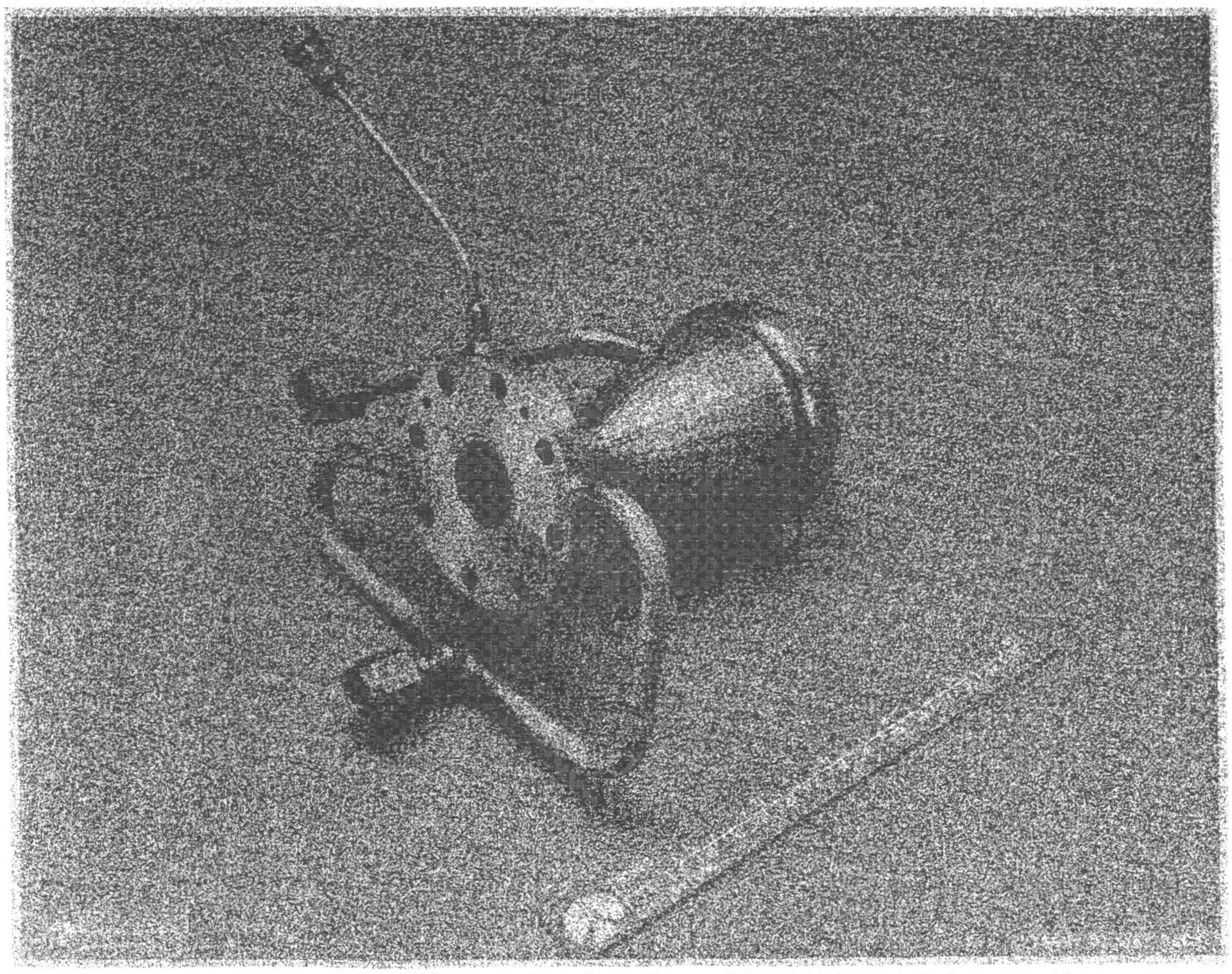


Figure 3-3. Thrust Chamber Assembly (15521-8/5/87-C1C*)

The heated hydrogen is transported through the return tube through a fitting, which splits the flow for dual inlets into the hydrogen injector manifold, and is then injected into the combustion chamber through the fuel annulus in each of 12 injector elements.

The chamber inner liner (Figure 3-4) is machined from NARloy-Z, a high-strength copper alloy, and contains 30 coolant passages. These passages are 0.020 in. wide and 0.030 in. deep in the combustion chamber and throat areas. In the nozzle area, the channel width is increased to 0.060 in. and the height transitions from 0.030 in. in the throat area to 0.060 in. in the nozzle area. See Appendix A Part No. 7R033603. The open channels in the liner are closed out by electrodepositing an outershell of nickel over the NARloy-Z liner. The coolant passages are filled with a wax prior to copper plating and electroforming of the nickel. The wax is removed after electroforming to produce the hydrogen coolant passages. A layer (0.003 to 0.005 in.) of copper is deposited prior to the nickel to prevent hydrogen embrittlement of the nickel during thruster operation. Prior to the closeout of the channels, the inlet flange/manifold and outlet manifolds (Figure 3-4) are brazed to the chamber liner.

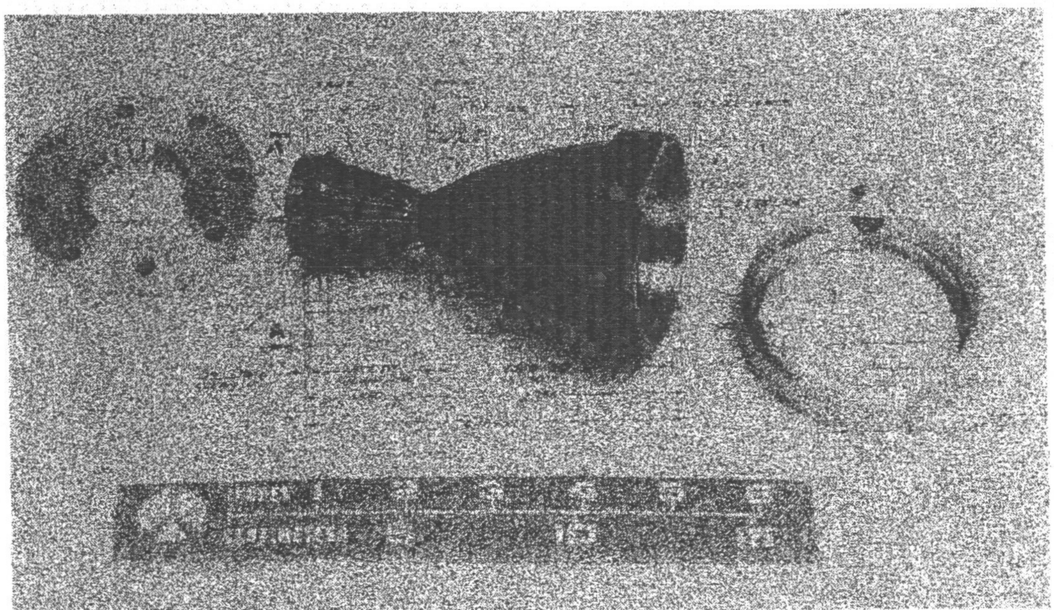


Figure 3-4. Thrust Chamber Components (1XZ91-4/30///85-C1C*)

3.1.1 Thrust Chamber Thermal Design and Predicted Life

A thermal analysis was conducted to define a coolant channel configuration for the LeRC thrust chamber that emulated the heat transfer of the prototype thrust chamber but with a reduced coolant channel pressure drop. A configuration was selected that increased the number of channels from 24 to 30 and reduced the channel cross section from 0.030 in. wide by 0.040 in. deep at the throat to 0.020 in. by 0.030 in. The predicted pressure drop, at a mixture ratio of 8 and a chamber pressure of 100 psi, was 46 psi. At a mixture ratio of 3, the predicted pressure drop was 97 psi (Figure 3-5). The predicted combustion gas side wall temperature profile at a mixture ratio of 8 with the redesigned coolant channels is shown in Figure 3-6(a). The measured prototype temperatures used to correlate and anchor the analysis are also indicated. The prediction was based on the measured total heat load to the hydrogen coolant. The maximum wall temperature, which occurs near the throat, was predicted to be 1120°F, which is acceptable for long life. The measured back wall temperature is also shown. The temperature profile within the thrust chamber NARloy-Z liner was predicted by computer analysis. A sample channel cross section is shown in Figure 3-6(b). The measured back wall temperature in the throat region is predicted to be approximately 75°F lower than the combustion gas side wall temperature (Figure 3-6(b)). Thermal conduction in the axial direction and boundary layer effects not included in the model tend to smooth out the actual temperature profile. The maximum wall temperature was predicted to be less than 800°F at a mixture ratio of 3 (Figure 3-7).

Figure 3-8 presents the projected NARloy-Z nozzle thermal fatigue characteristics and depicts cycle life in terms of full thermal cycles as a function of nozzle wall radial temperature differential. If the calculated value of wall temperature differential (ΔT) is used (75°F), the expected thrust chamber cycle life would exceed 100,000 full thermal cycles. Several seconds of firing time would be required to create a full thermal cycle. Short pulse tests would not be expected to create the maximum temperature differential.

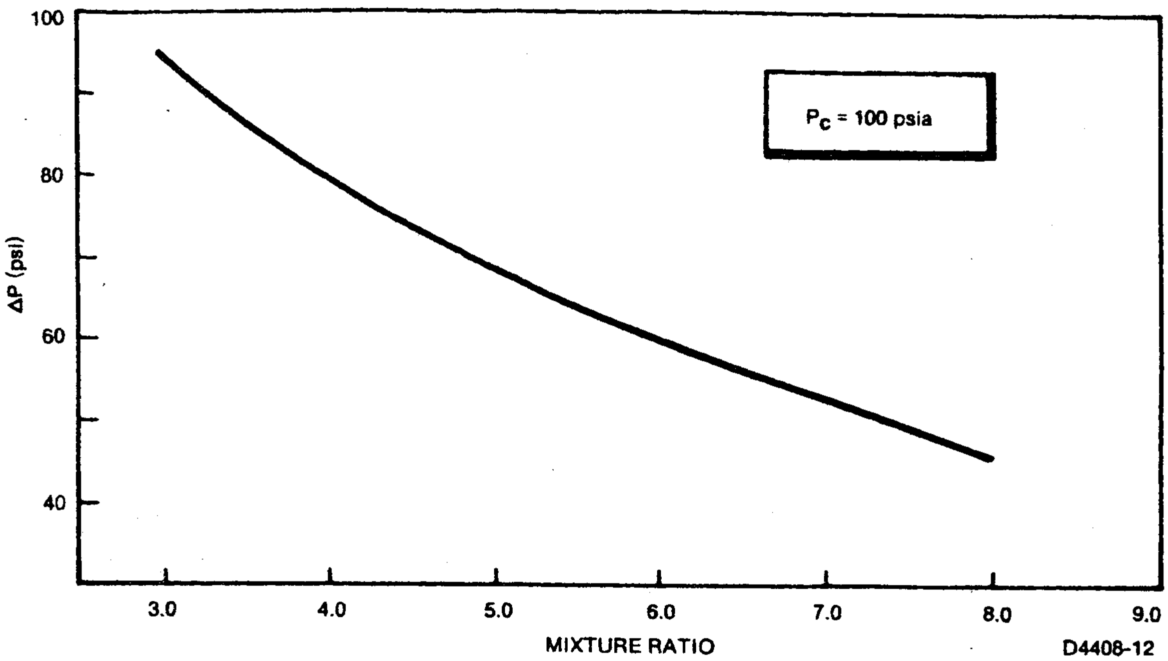


Figure 3-5. Predicted Pressure Drop in Redesigned Coolant Channels

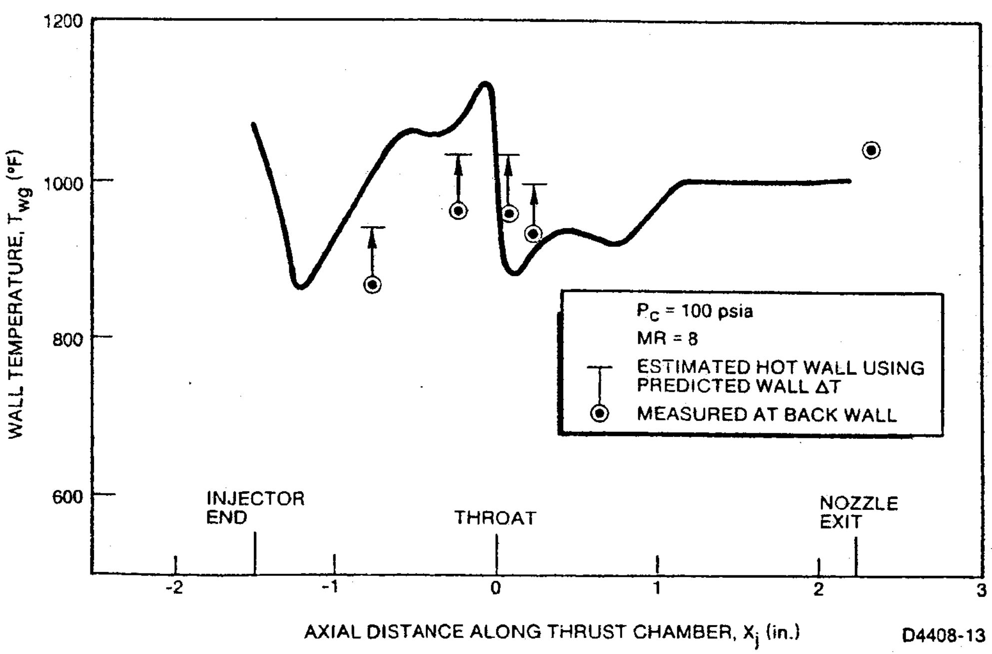


Figure 3-6.(a). Predicted Combustion Gas Side Wall Temperature

```
NUMBER OF ITERATIONS =  899
DIFFERENCE BETWEEN HEAT IN AND HEAT OUT =  .03057 PERCENT

HEAT INFLUX =    5.154
2-D/1-D Q/A =  .97783
```

HOT GAS WALL

1121	1121	1121	1121	1121	1121
1106	1106	1107	1107	1107	1107
1092	1092	1092	1092	1092	1093
1077	1077	1077	1079	1080	1080
1073	1073	1071			
1069	1068	1066			
1066	1065	1061		COOLANT	
1063	1061	1058		CHANNEL	
1060	1058	1054			
1058	1056	1052			
1056	1054	1049			
1055	1053	1046	1039	1035	1034
1048	1048	1047	1046	1046	1045
1047	1047	1047	1047	1047	1047

BACK SIDE WALL

```
    1.  LAND WIDTH =  .03440
    2.  CHANNEL WIDTH =  .02000
    3.  WALL THICKNESS =  .04000
    4.  CHANNEL DEPTH =  .03000
    5.  CLOSEOUT THICKNESS =  .04000
    6.  TAW = 4501. DEG. F
    7.  HG =  .0015246
    8.  TC =  639. DEG. F
    9.  REFERENCE HC =  .0079705
   10.  HC FACTOR FOR UPPER WALL = 1.0000
   11.  HC FACTOR FOR LOWER WALL = 1.0000
   12.  EXPONENT =  .5500
13-14.  K OF REGION 1 =  .004774 + ( 0.          ) * T
15-16.  K OF REGION 2 =  .004774 + ( 0.          ) * T
17-18.  K OF REGION 3 =  .000900 + ( 0.          ) * T
   19.  CONVERGENCE CRITERION =  .0010 DEG. F
   20.  COATING THICKNESS = 0.000000
21-22.  COATING K = 0.         + ( 0.          ) * T
```

D4408-21

Figure 3-6(b). Predicted Thrust Chamber Liner Temperatures

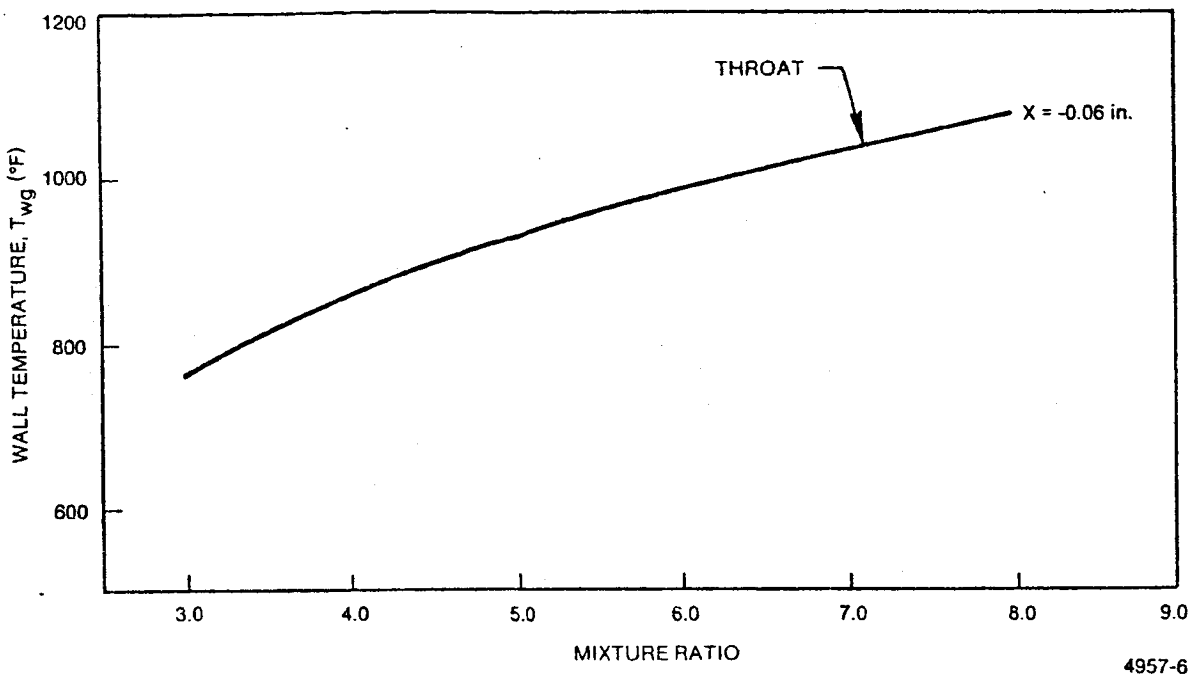


Figure 3-7. Predicted Effect of Mixture Ratio on Combustor Wall Temperature

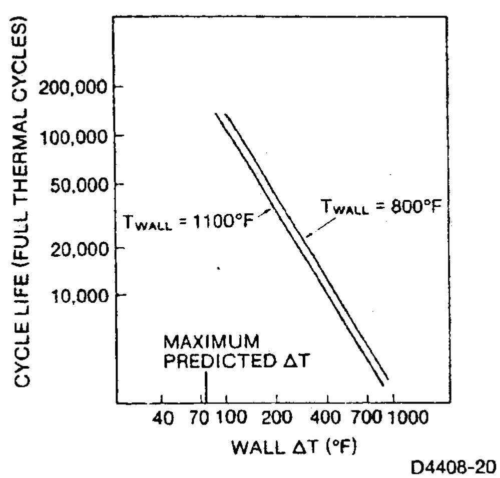


Figure 3-8. Projected Thrust Chamber Cycle Life

3.1.2 Expansion Area Ratio Effects on Chamber Coolant Temperatures

The prototype and LeRC thrust chamber expansion area ratio of, ϵ, 30:1 was chosen as a compromise between specific impulse, thruster temperature, and test facility vacuum pumping capability. The test results clearly indicate that an increase in expansion area ratio could be realized for flight-type hardware. This is particularly true for the low-heat-flux injector, since it lowers the hardware temperature and increases design margins significantly. An increase in expansion ratio can be accomplished by adding a radiation-cooled expansion skirt, by extending the cooled portion of the nozzle, or a combination of both. The hardware temperature at the attachment point of an uncooled skirt decreases as the expansion area increases. This type of hardware design would involve tradeoffs to be performed considering the temperature at the attach point, materials to be used for the uncooled skirt, and the details of the attachment point. To provide some insight for design tradeoffs, the increase in coolant temperature was approximated for increases in expansion ratio beyond 30. Figure 3-9 displays the results of the calculations.

3.1.3 Thrust Chamber Fabrication

External skin temperature circumferential variations and the higher-than-predicted thrust chamber flow pressure drop observed during testing raised questions concerning the coolant passage dimensions. Flow tests using hot and cold water and infrared cameras (similar to procedures used for the Space Shuttle Main Engine [SSME]) did not show any blocked passages. No other nondestructive inspection method was readily available for verifying channel-by-channel dimensions along the length of the thrust chamber.

A spare thrust chamber liner was available that had not had the nickel electroform closeout completed. The channels were open for detail inspection. The flange and exit manifold had been brazed to the liner. A detailed channel-by-channel dimensional inspection was made on this spare liner. Three discrepant characteristics were found: variable channel depth and two

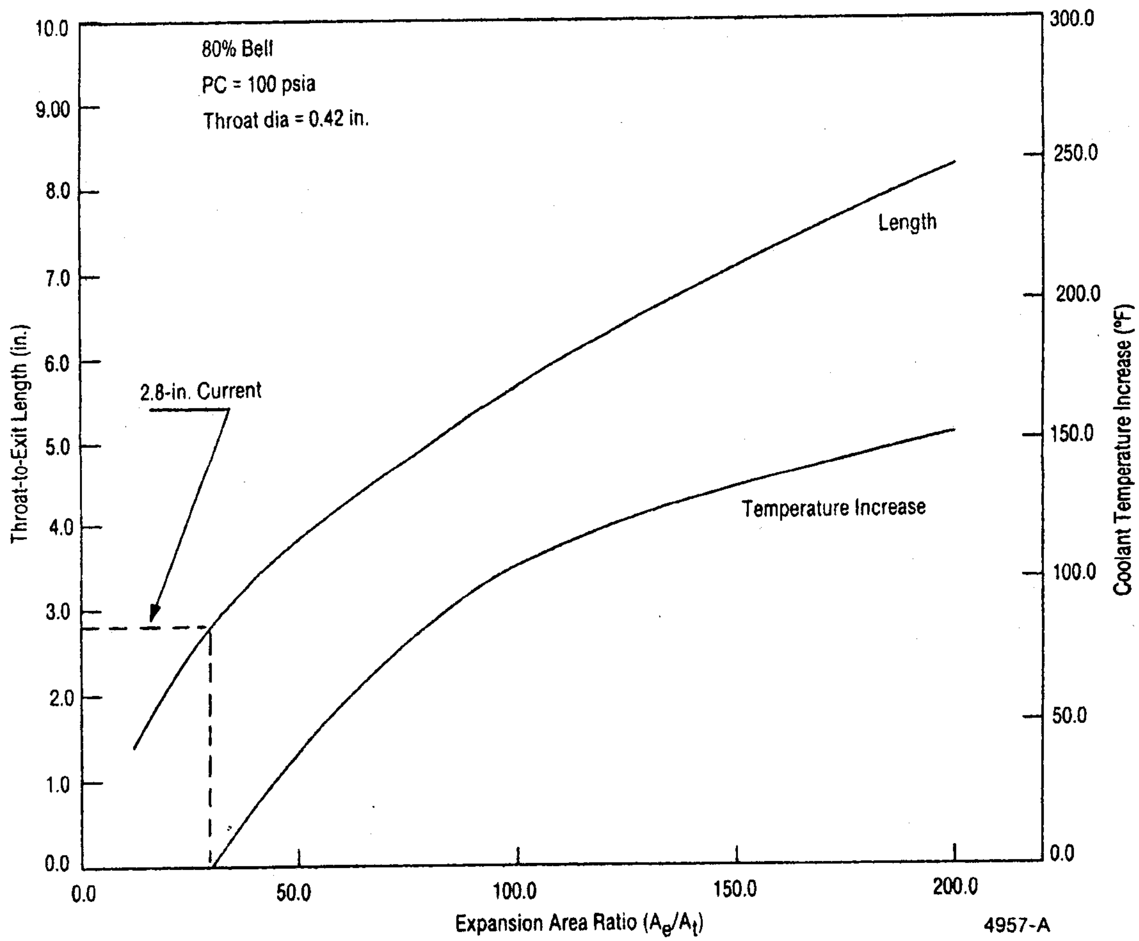


Figure 3-9. Expansion Area Ratio Effects on Coolant Temperatures

discontinuities in the bottom of the channel caused by improper programming of the digital-controlled machine by the vendor that milled the channels in the liner. The results are shown in Figure 3-10.

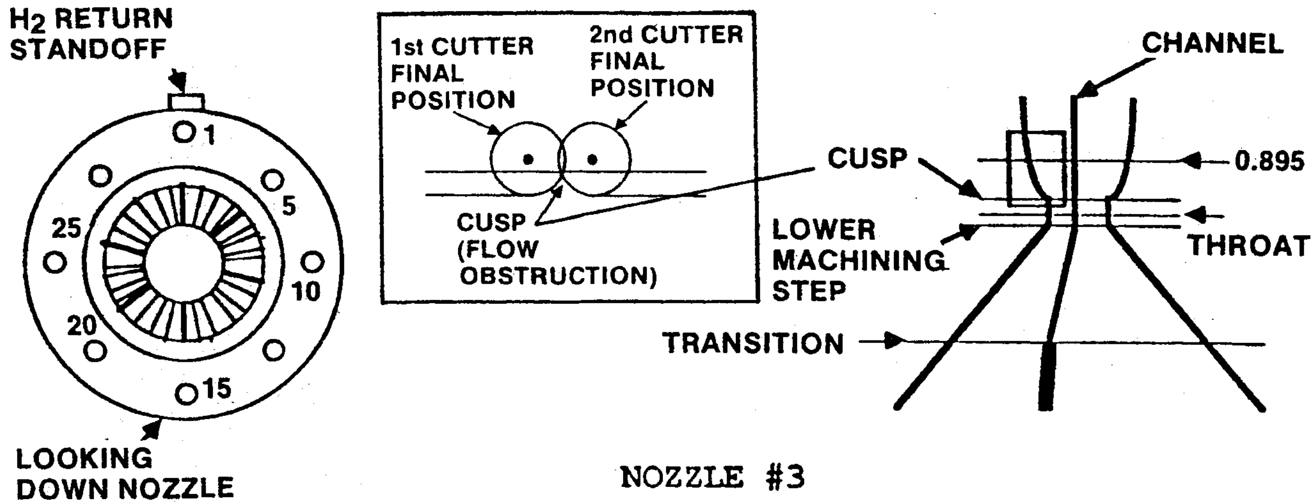
The discrepancy in the combustion section was a lip or cusp in the bottom of the channel caused by a nonoverlap of starting and stopping the milling cutter used to machine the channels. The lack of cutter start and stop center overlap caused a small piece of metal to be left in the channel bottom, which obstructed the hydrogen coolant flow. Also, a machining step immediately downstream of the throat section apparently caused by a milling machine programming error at the transition from the throat radius to the bell contour of the nozzle. The effects of this step on the thrust chamber coolant flow, while not desirable, were considered minimal.

The variable channel-to-channel depth was apparently caused by off-center tooling, which presented the nozzle blank in a concentric manner to the programmed, moving, milling cutter.

The cusp and the machining step were built into the fabrication computer program; therefore, their presence in LeRC 1 and LeRC 2 thrusters is ensured. The presence or absence of the variable channel depth cannot be verified, nor can its circumferential relationship to the nozzle flanges be verified.

The cusp reduces the coolant flow area by 30 to 50%, which has a marked effect on the nozzle flow pressure drop and could cause sonic flow to take place at the cusp. The high pressure drop and circumferential variable temperatures can be explained by these effects but not quantified. The increased coolant channel pressure drop requires that a correspondingly higher inlet pressure be supplied to the thruster to maintain chamber pressure at desired levels.

A full-length channel-by-channel inspection is recommended for the fabrication of any future units.



NOZZLE #3

CHANNEL	LOWER MACHINING STEP (.030)	THROAT (.030)	CUSP (.030)	0.895 (.030)	WIDTH (.020)
1	.0365	.0320	.0180	.0349	.02025
2		.0326			.02000
3		.0282			.02020
4		.0298			.02025
5	.0325	.0254	.0145	.0274	.02050
6		.0267			.02000
7		.0256			.02005
8		.0229			.02005
9		.0221			.02025
10	.0258	.0171	.0083	.0170	.02000
11		.0206			.02070
12		.0125			.02040
13		.0147			.02000
14		.0131			.02000
15	.0243	.0173	.0065	.0100	.02005
16		.0109			.02005
17		.0177			.02030
18		.0162			.02010
19		.0162			.02005
20	.0297	.0211	.0150	.0195	.02005
21		.0163			.02010
22		.0230			.02000
23		.0279			.02025
24		.0275			.02025
25	.0360	.0298	.0190	.0292	.02020
26		.0288			.02030
27		.0303			.02015
28		.0253			.02020
29		.0307			.02030
30		.0306			.02015

4957-38

Figure 3-10. 25-lb Thruster Nozzle Channel Dimensions

3.2 INJECTOR

Figure 3-11 shows the injector prior to assembly and braze. Six of the oxidizer posts have been inserted into the injector body for illustrative purposes. The injector body components and oxidizer posts were fabricated from 316L SS bar stock. The tube components were fabricated from 321 SS tubing. The injector faceplate is NARloy-Z. The igniter/spark plug is located at the center of the injector. Nine percent of the oxidizer flow is introduced into the igniter cavity by two 0.030-in. orifices drilled into the oxidizer inlet manifold. The remaining oxidizer is introduced into the combustor through the 12 oxidizer posts. These posts are located in the center of the 12 coaxial elements and are recessed 0.060 in. from the injector face. The oxidizer igniter flow is surrounded by GH_2, introduced through twelve 0.016-in. orifices that provide like-on-like impinging streams. The combustor uses hydrogen for boundary layer coolant (BLC). The BLC is introduced through twelve 0.039-in. showerhead orifices located at the perimeter of the injector. The primary hydrogen flow is injected into the combustion chamber through the annulus formed by the outside diameter of the oxidizer post and the hydrogen orifice wall. This coaxial element mixes and distributes combustible gases into the combustion chamber, providing a flow field consisting of an oxygen core surrounded by a hydrogen annulus. Table 3-2 displays the flow distribution to various injector distribution elements. Figure 3-12 shows a face-on view of the injector and indicates the propellant injection features.

3.2.1 Injector Fabrication

To verify flow areas and characteristics, the injector element dimensions were measured after assembly. Table 3-3 summarizes the results of the work and presents the drawing tolerance limits for reference.

The control of the fuel annulus gap variations and the concentricity of the fuel annulus to the oxidizer post outside diameter could have been improved. These variations probably contributed to the circumferential

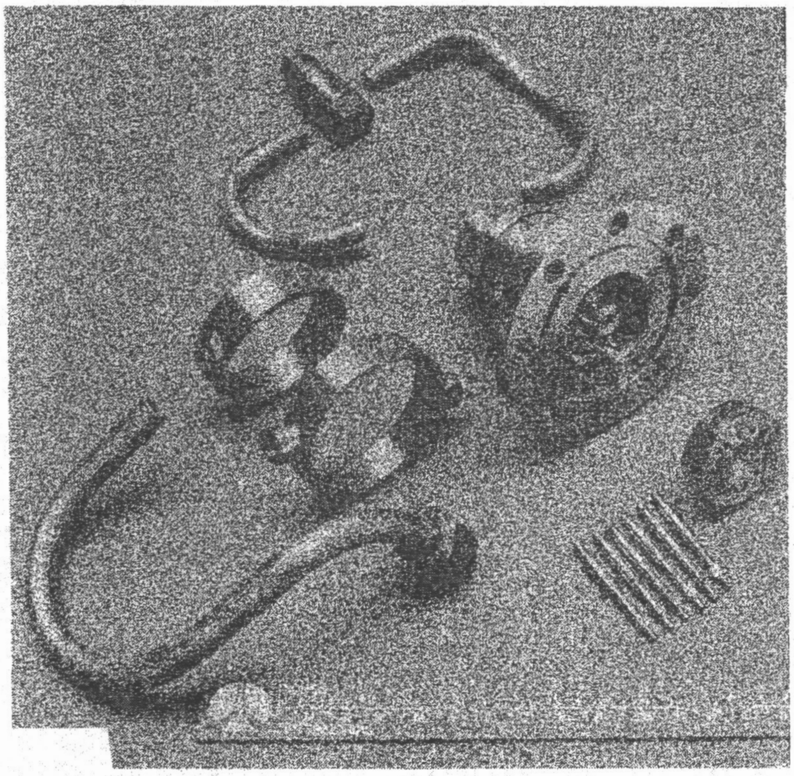
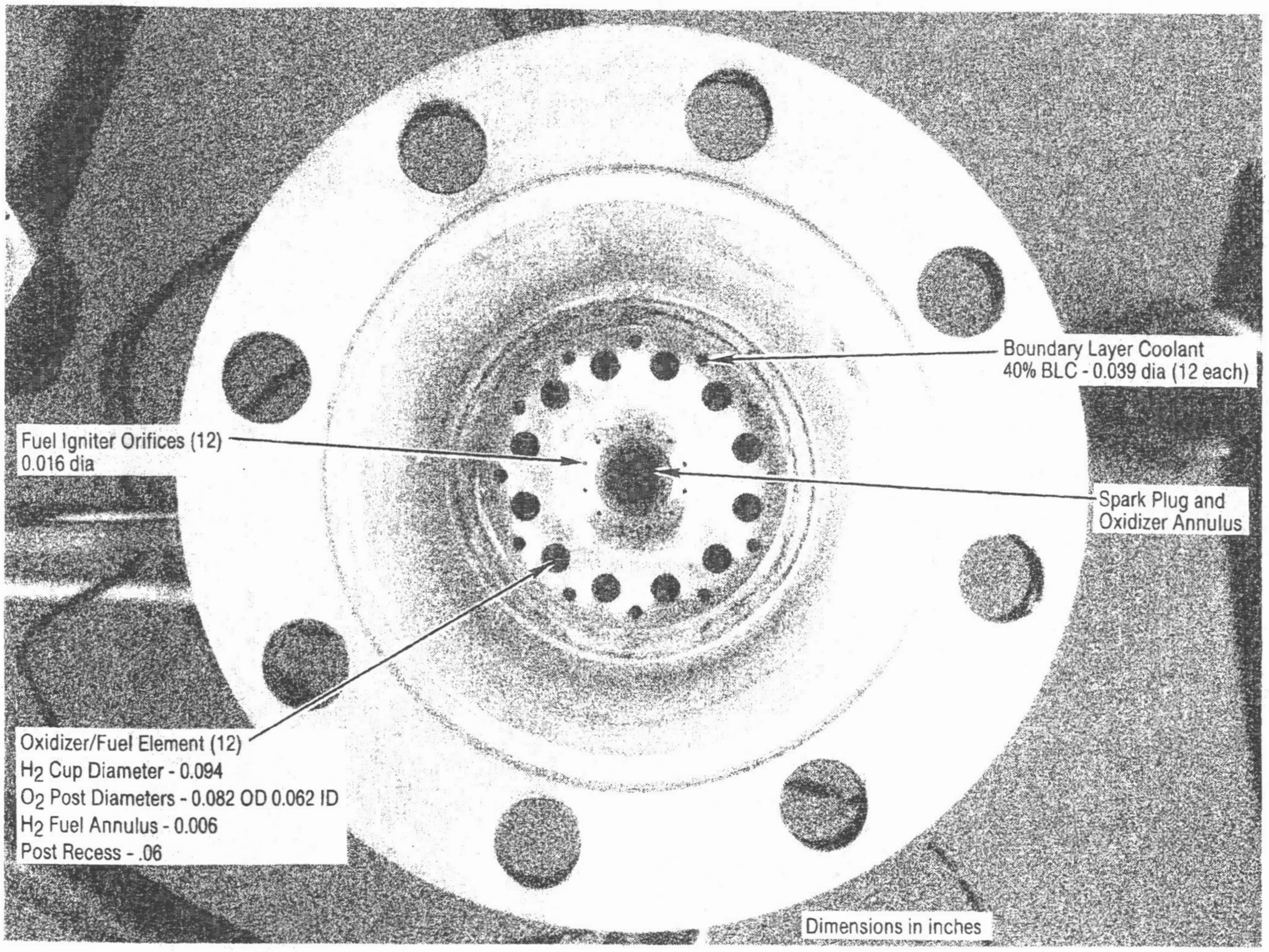


Figure 3-11. GO_2/GH_2 Injector Assembly Layout (15521-8/8/87-C1B*)

Table 3-2. Injector Flow Distribution

Element		Flow Area (%)
Oxidizer		
Igniter	0.30-in. orifice (2)	9.0
Coaxial	0.060-in. ID posts (12)	91.0
Fuel		
Igniter	0.016-in. orifices (12)	6.6
BLC	0.039-in. orifices (12)	39.2
Coaxial	0.094-in. OD x 0.082-in. ID annulus (12)	54.2
Igniter mixture ratio		13.4
Coaxial element mixture ratio		10.9
Overall mixture ratio		8.0



4957-9

Figure 3-12. LeRC 25-lbf GO_2/GH_2 Injector (1XZ25-10/28/75-C1B*)

Table 3-3. Injector Element Dimensions (After Assembly)

Injector	Fuel Annulus Gap (in.)		Concentricity of Fuel Annulus to Oxygen Post (in.)		
	Minimum	Maximum	Minimum	Maximum	Average
LeRC 1	0.0032	0.0076	0.0003	0.0026	0.0019
LeRC 2	0.0032	0.0113	0.0016	0.0039	0.0029
Low-heat-flux	0.0045	0.0098	0.0005	0.0036	0.0019
Drawing requirements	0.005	0.007	0.000	0.003	NA
Recommended	0.0055	0.0065	0.0000	0.0006	NA

variations in the thrust chamber external nozzle skin temperatures observed during testing. The testing pointed out the need for improved (tighter) tolerances to reduce the flow variations in the injector. Any units fabricated in the future should have reduced dimensional tolerances concerned with flow area variations and the oxidizer post concentricity to the fuel annulus. Recommended values are included in Table 3-3.

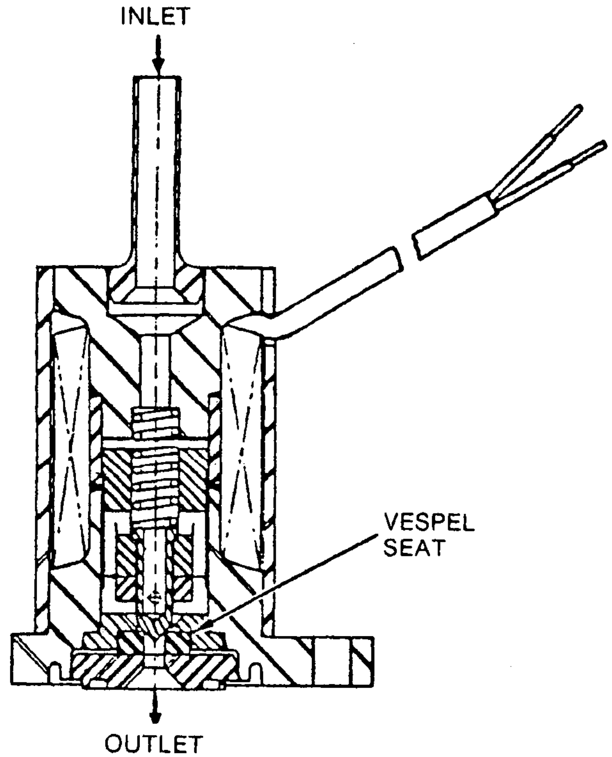
3.3 VALVES

The thruster incorporates separate, identical fuel and oxidizer valves. The valves (Figure 3-13) are manufactured by Wright Components (P/N 18001-11). These valves are a direct-operated, normally closed, spring-return, coaxial solenoid valves. The valves operate on $\pm$28 Vdc. These valves have proven to be very reliable during extensive endurance and pulse mode testing of the prototype and LeRC thrusters.

3.4 IGNITION SYSTEM

Conventional electrical high-voltage spark ignition systems were used throughout the program. Three systems (i.e., [SSME, Simmonds Precision, and J-2]) were used as summarized in Table 3-4. All systems used the Simmonds Precision spark plug (Figure 3-14). The SSME-type system used the SSME qualified exciter (Figure 3-15) threaded directly to the spark plug. This approach is planned to be applied to the flight hardware for the Freedom Station.

Prior to the contract award, two SSME exciters were modified to increase the spark rate from 75 sparks/s to 225 sparks/s. This modification was made to accommodate the anticipated minimum thrust pulse duration of 30 ms. The modification to the SSME exciters was improperly fabricated (or installed) and the exciter units would not perform. Subsequent studies have shown that a pulse duration approximating 250 ms is adequate, and the qualified SSME producing 75 sparks/s satisfies the need without modification.



VENDOR	WRIGHT COMPONENTS, INC.
PART NUMBER	18001
TYPE	COAXIAL SOLENOID
HISTORY	CRYOGENIC VALVES FOR GENERAL DYNAMICS CENTAUR PROGRAM
SEALING	
LEAKAGE	<10 SCC/MIN HELIUM AT TEMPERATURE EXTREMES
MATERIALS	METAL POPPET/VESPEL SEAT
SOLENOID PLUNGER MATERIALS	• AISI 430 STEEL • NITRONIC 60 WEAR RING
SOLENOID	
MATERIALS	• AISI 430 STEEL FOR MAGNETIC PARTS • AISI 304L STEEL FOR NONMAGNETIC PARTS
ASSEMBLY	ELECTRON-BEAM WELDED
COIL DESIGN	• 2100 TURNS OF COPPER MAGNET WIRE • VACUUM IMPREGNATED AFTER WINDING WITH EMERSON AND CUMINGS SYCAST 2651 EPOXY

4957-10

Figure 3-13. Weight Components Propellant Valve

Table 3-4. Spark Igniter Systems

System	Output Voltage (kV)	Input Voltage (V)	Spark Energy (MJ/Spark)	Spark Rate (Hz)
SSME (Modified)*	8	20-24	12	225
Simmonds	6.8	10-30	250	60
J-2	20-32	24-30	90	50

*Modified from 75 sparks/s for the qualified SSME igniter.

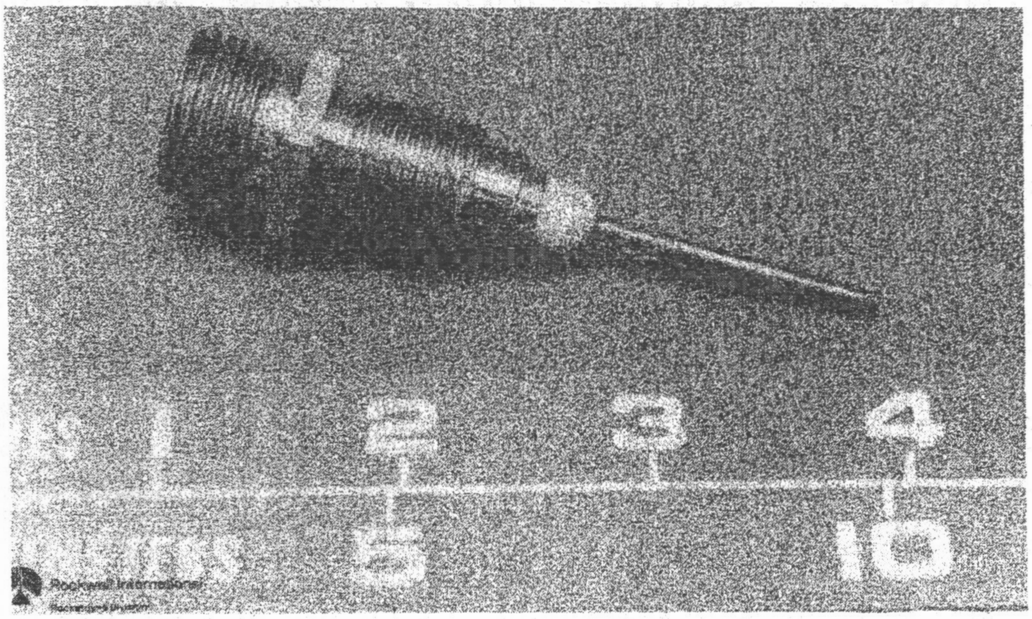
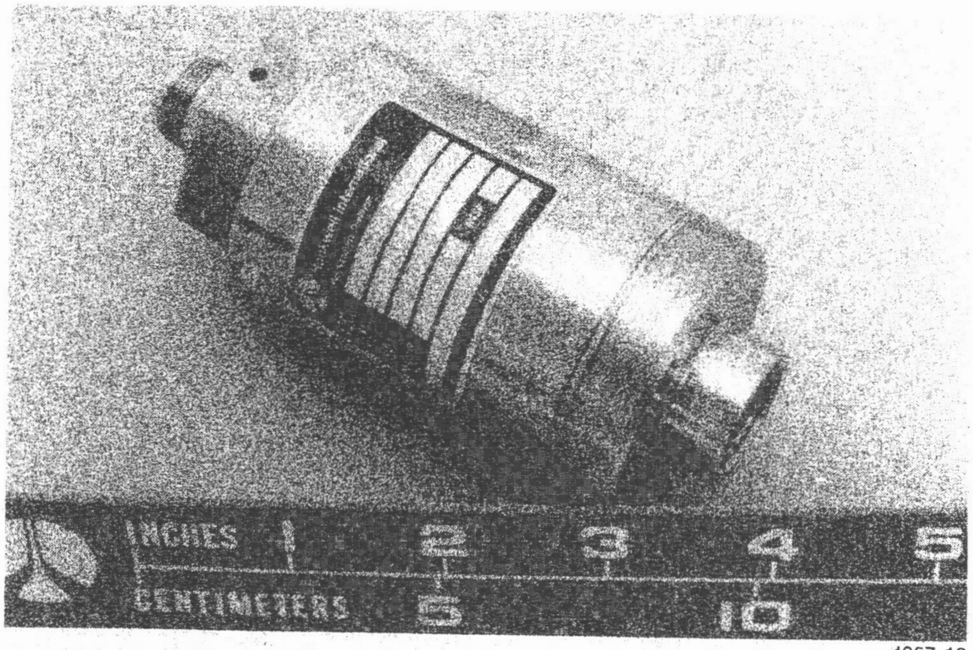


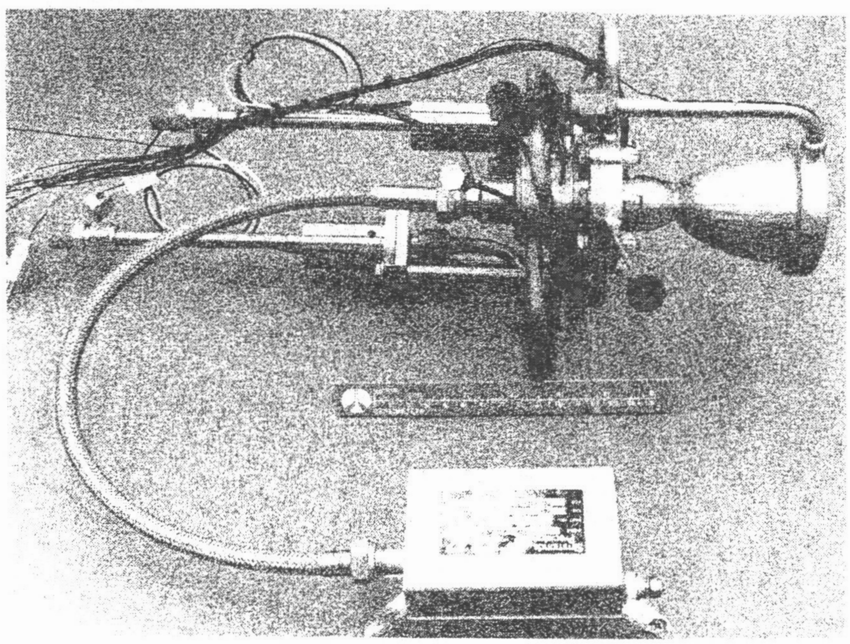
Figure 3-14. Simmonds Precision Spark Plug (SC87D-13-296)

An aircraft-type exciter with spark cable was available from Simmonds Precision, and two units were ordered for use in the program. In Figure 3-16, the Simmonds Precision unit is shown with the cable (without the pressurizing sleeve) assembled to the thruster. These units worked only sporadically in the high-vacuum test firing chamber of test stand 302 at MSFC.



4957-12

Figure 3-15. SSME Spark Exciter (86D-9-701)



4957-13

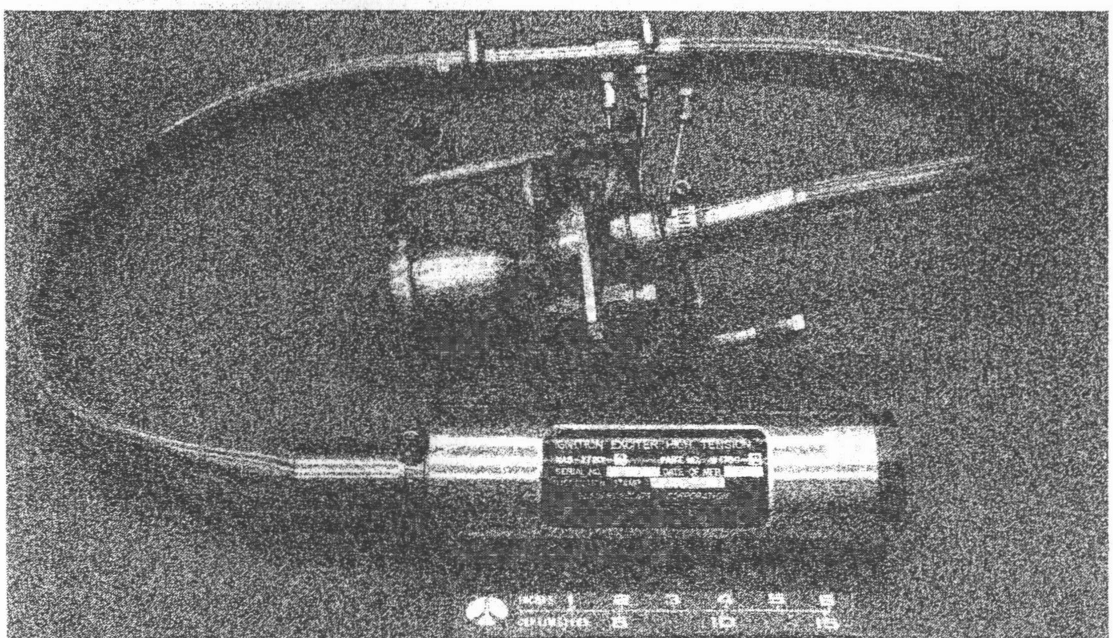
Figure 3-16. LeRC 1 Thruster with Simmonds Exciter and Cable (14421-8/20/87-C1A*)

The J-2 exciter from the Rocketdyne J-2 engine had been used successfully during the prototype testing and pulsing in the same vacuum facility. The J-2 exciter, cable, and pressurizing sleeve (Figure 3-17) were employed after the Simmonds Precision units proved unsuccessful at the vacuum condition. To prevent arcing, the spark cable was jacketed by a sleeve containing atmospheric ambient pressure. The sleeve extended from the remotely mounted exciter to the spark plug. No further problems were encountered except when, on occasion, the pressure integrity of the cable jacket was inadvertently compromised.

3.5 INSTRUMENTATION

Both the injector and chamber were fabricated with pressure taps, with 1/8-in. tubes brazed into these taps for ease of interfacing. The taps are located in the hydrogen chamber inlet manifold, hydrogen injector inlet manifold, oxygen injector inlet manifold and at the head end of the spark plug bore for chamber pressure. (See Appendix A, Drawing 7R033657 Section A-A and 7R033603.) Marotta transducers were used for pressure measurement.

Internal gas temperatures of the hydrogen and oxygen manifolds were measured with inconel type K thermocouples (1/16-in. sheath) inserted through the pressure tubes and into the manifold flow field. Pressures were measured through a tee fitting used to install and retain the thermocouples in the tube. The thruster external temperatures were measured with chromel-alumel type K thermocouples which were spot-welded to their respective positions on the skin of the injector and chamber.



4957-14

Figure 3-17. LeRC 2 Thruster, J-2 Exciter, Cable, and Pressurizing Sleeve (15561-9/1/88-C1*)

4.0 HOT-FIRE TESTING

The thruster was tested in conjunction with the Freedom Station propulsion test bed program. The test bed installed in the 20-ft-diameter altitude test cell 302 at MSFC (Figure 4-1) is designed to be representative of a space station propulsion system and consists of the propulsion module, propellant storage module, and electrolysis module. These modules can be operated individually or in combination with each other. In this program, the thruster was operated in conjunction with the propellant storage module. The thruster is shown installed in the propulsion test bed (Figure 4-2), and the test log, summarizing the tests conducted and objectives, the hardware used, the test conditions and results, and remarks (as applicable) for each test, are presented in Table 4-1.

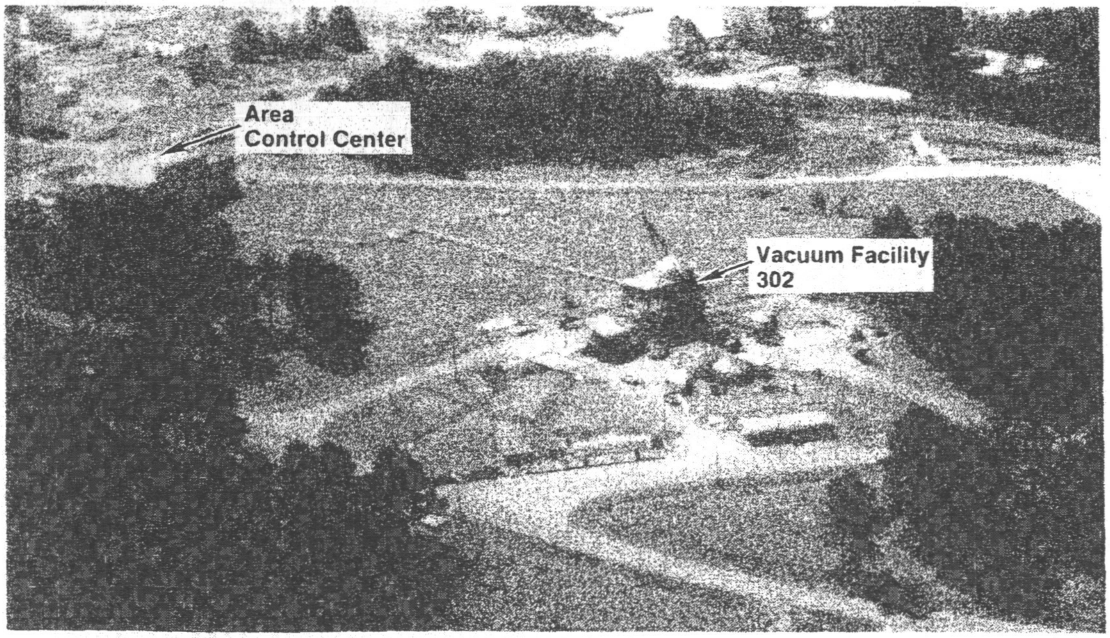


Figure 4-1. Vacuum Facility 302 (86D-9-706)

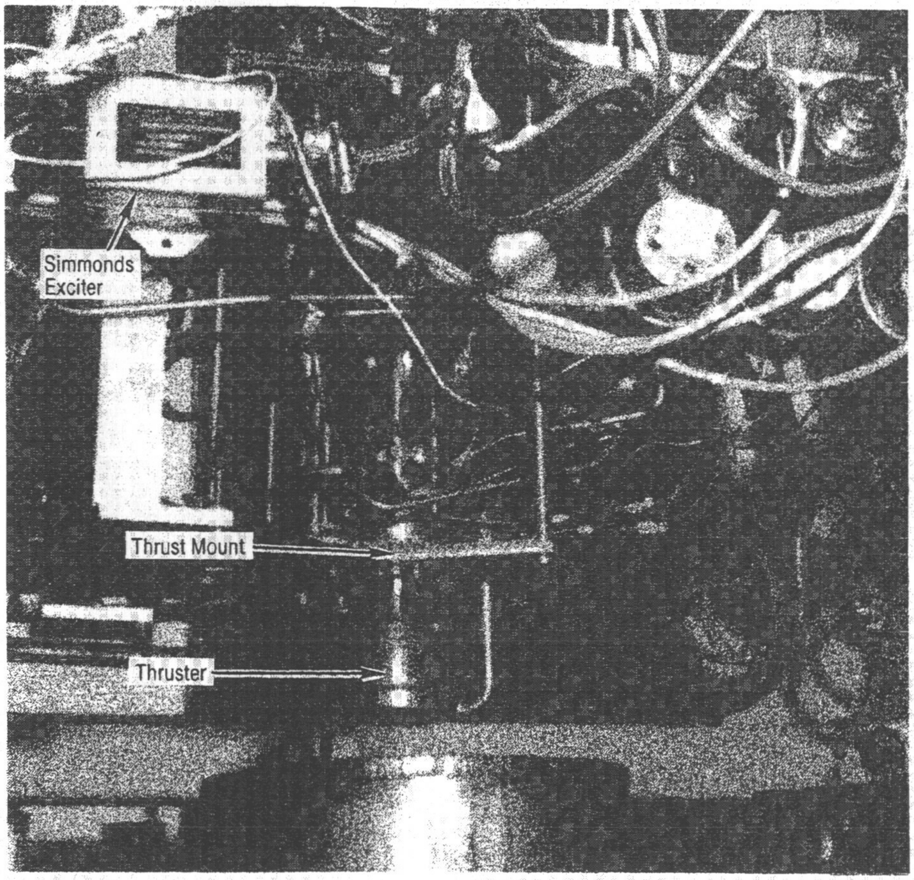


Figure 4-2. Thruster Installation in Propulsion Test Bed

Table 4-1. GO_2/GH_2 25-lbf Thruster Test Log--8/26/88
(Sheet 1 of 3)

TEST NUMBER	DATE	TEST OBJECTIVE	HARDWARE CONFIGURATION: INJECTOR	HARDWARE CONFIGURATION: CHAMBER	HARDWARE CONFIGURATION: IGNITER	TEST CONDITIONS AND STEADY STATE: MIXTURE RATIO (O/F)	TEST CONDITIONS AND STEADY STATE: Pcns (psia)	DURATION: SCHEDULE (sec)	DURATION: ACTUAL (sec)	TEST RESULTS	REMARKS
P103-043	09/01/87	SYSTEM BLOWDOWN	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	N/A	N/A	N/A	INERT BLOWDOWN TEST
P103-044	09/01/87	SYSTEM BLOWDOWN	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	N/A	N/A	N/A	INERT BLOWDOWN TEST
P103-045	09/01/87	SYSTEM BLOWDOWN	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	N/A	N/A	N/A	INERT BLOWDOWN TEST
P103-046	09/01/87	HOT FIRE CHECKOUT	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	1	1	IGN	SUCCESSFUL CHECKOUT TEST
P103-047	09/02/87	HOT FIRE CHECKOUT	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	10	0	NO IGN	
P103-048	09/02/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	10	0	NO IGN	
P103-049	09/02/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	10	0	NO IGN	
P103-050	09/11/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	10	0	NO IGN	
P103-051	09/11/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	10	0	NO IGN	
P103-052	09/15/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	5	5	IGN	MIRROR INSTALLED, IGNITION CABLE PRESSURIZED
P103-053	09/15/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	2	0	NO IGN	MIRROR INSTALLED, IGNITION CABLE PRESSURIZED
P103-054	09/15/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	2	2	IGN	MIRROR, PRESS. CABLE, CHANGED SEQUENCE
P103-055	09/15/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	2	0	NO IGN	H2 PROP VALVE FAILED
P103-056	09/17/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	10	0	NO IGN	OLD SEQ, NO MIRROR
P103-057	09/17/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS,01	8.05	102.27	10	10	IGN	NEW SEQUENCE, NO MIRROR
P103-058	09/17/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	30	0	NO IGN	
P103-059	09/17/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS,01	8.00	100.00	30	0	NO IGN	DG TIMING BLIP
P103-060	09/29/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS, 02	8.00	100.00	30	0	NO IGN	CHANGED OUT IGNITION SYSTEM- CABLE AT ATM
P103-061	09/29/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS, 02	8.00	100.00	30	0	NO IGN	NO PR6
P103-062	09/29/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS, 02	8.00	100.00	30	0	NO IGN	
P103-063	09/30/87	IGNITER INVESTIGATION	LeRC#1	LeRC#1	SIMONDS, 02	8.00	100.00	1	0	NO IGN	AT ATM PRESS., WITH MIRROR
P 103-065	10/01/87	IGNITER INVESTIGATION	PROTOTYPE	PROTOTYPE	J-2	8.00	100.00	1	0	NO IGN	AT ATM PRESS., WITH MIRROR
P103-066	10/08/87	J-2 IGNITION CHECKOUT	PROTOTYPE	PROTOTYPE	J-2	8.00	100.00	1	1	IGN	SUCCESSFUL J-2 IGNITION SYSTEM TEST
P103-067	10/08/87	J-2 IGNITION CHECKOUT	PROTOTYPE	PROTOTYPE	J-2	8.00	100.00	1	1	IGN	SUCCESSFUL IGNITION REPEATIBILITY TEST
P103-068	10/08/87	J-2 IGNITION CHECKOUT	PROTOTYPE	PROTOTYPE	J-2	8.00	100.00	1	1	IGN	SUCCESSFUL IGNITION REPEATIBILITY TEST
P103-069	10/08/87	J-2 IGNITION CHECKOUT	PROTOTYPE	PROTOTYPE	J-2	8.00	100.00	1	1	IGN	SUCCESSFUL IGNITION REPEATIBILITY TEST
P103-070	10/08/87	J-2 IGNITION CHECKOUT	PROTOTYPE	PROTOTYPE	J-2	8.00	100.00	1	1	IGN	SUCCESSFUL IGNITION REPEATIBILITY TEST
P103-071	10/09/87	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	7.71	102.30	10	10	IGN	SUCCESSFUL LeRC#1 DEMONSTRATION TEST
P103-072	10/09/87	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	7.61	99.87	30	30	IGN	TEMP R/L CUT
P103-073	10/09/87	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	6.02	111.01	30	30	IGN	TEMP R/L CUT
P103-074	10/15/87	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	6.08	112.65	120	47	IGN	TEMP R/L CUT
P103-093	12/16/87	PERFORMANCE/COMPATABILITY DEMO	LHF, 40% BLC	LeRC#1	J-2	8.00	100.00	2	2	IGN	SUCCESSFUL LHF INJ. 40% BLC CHECKOUT TEST
P103-094	12/16/87	PERFORMANCE/COMPATABILITY DEMO	LHF, 40% BLC	LeRC#1	J-2	8.06	95.64	10	10	IGN	SUCCESSFUL PERFORMANCE/COMPATIBILITY TEST
P103-095	12/16/87	PERFORMANCE/COMPATABILITY DEMO	LHF, 40% BLC	LeRC#1	J-2	8.02	97.34	30	30	IGN	SUCCESSFUL PERFORMANCE/COMPATIBILITY TEST
P103-096	12/16/87	PERFORMANCE/COMPATABILITY DEMO	LHF, 40% BLC	LeRC#1	J-2	8.00	97.43	120	120	IGN	SUCCESSFUL PERFORMANCE/COMPATIBILITY TEST
P103-097	01/06/88	HOT FIRE CHECKOUT	LHF, 0% BLC	LeRC#1	J-2	8.00	100.00	2	0	NO IGN	LOW HEAT FLUX INJECTOR, 0% BLC, VAC ON CABLE
P103-098	01/06/88	HOT FIRE CHECKOUT	LHF, 0% BLC	LeRC#1	J-2	8.00	100.00	2	0	NO IGN	LOW HEAT FLUX INJECTOR, 0% BLC, 2 PSI ON CABLE
P103-099	01/06/88	HOT FIRE CHECKOUT	LHF, 0% BLC	LeRC#1	J-2	8.00	100.00	2	0	NO IGN	LOW HEAT FLUX INJECTOR, 0% BLC, 5 PSI ON CABLE
P103-100	01/12/88	HOT FIRE CHECKOUT	LHF, 0% BLC	LeRC#1	J-2	8.00	100.00	2	2	IGN	SUCCESSFUL HOT FIRE CHECKOUT TEST
P103-101	01/12/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 0% BLC	LeRC#1	J-2	8.25	109.08	10	10	IGN	SUCCESSFUL TEST
P103-102	01/13/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 0% BLC	LeRC#1	J-2	7.97	106.82	30	15	IGN	LOW HEAT FLUX INJECTOR, 0% BLC, TEMP R/L CUT
P103-126	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	8.00	100.00	2	2	IGN	SUCCESSFUL LeRC#1 CHECKOUT TEST
P103-127	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	8.18	105.56	10	10	IGN	SUCCESSFUL REPEAT HOT FIRE TEST
P103-128	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	8.00	100.00	120	0	NO IGN	GOX VALVE DID NOT OPEN-INLET PRESSURE TOO HI
P103-129	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	8.15	107.90	120	28	IGN	HIGH TEMP CUT
P103-130	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	7.02	110.70	120	26	IGN	HIGH TEMP CUT
P103-131	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	7.25	150.00	120	0	NO IGN	GOX VALVE DID NOT OPEN-INLET PRESSURE TOO HI
P103-132	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	7.25	150.00	120	0	NO IGN	GOX VALVE DID NOT OPEN-INLET PRESSURE TOO HI
P103-133	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	7.28	139.14	120	39	IGN	INCREASED THRUSTER VOLTAGE FROM 30V TO 37V
P103-134	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	8.25	130.46	120	32	IGN	HIGH TEMP CUT
P103-135	07/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC#1	LeRC#1	J-2	4.85	118.88	120	120	IGN	SUCCESSFUL DURATION TEST

Table 4-1. GO_2/GH_2 25-lbf Thruster Test Log--8/26/88
(Sheet 2 of 3)

TEST NUMBER	DATE	TEST OBJECTIVE	HARDWARE CONFIGURATION INJECTOR	CHAMBER	IGNITER	TEST CONDITIONS AND STEADY STATE MIXTURE RATIO (O/F)	Pcns (psia)	DURATION SCHEDULE (sec)	ACTUAL (sec)	TEST RESULTS	REMARKS
P103-136	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	5.92	111.82	120	22	IGN	HIGH TEMP CUT
P103-137	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	4.99	87.80	120	27	IGN	HIGH TEMP CUT
P103-138	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	5.00	57.44	120	29	IGN	HIGH TEMP CUT
P103-139	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	7.34	51.92	120	20	IGN	HIGH TEMP CUT
P103-140	02/17/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	8.07	104.64	120	24	IGN	HIGH TEMP CUT
P103-141	02/19/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	8.16	107.17	120	28	IGN	HIGH TEMP CUT
P103-142	02/19/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	5.31	118.86	120	120	IGN	
P103-143	02/19/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	5.89	144.98	120	120	IGN	
P103-144	02/19/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	4.14	93.00	120	17	IGN	HIGH CHAMBER PRESS CUT
P103-145	02/19/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	4.11	93.66	120	120	IGN	
P103-146	02/19/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	6.97	80.13	120	17	IGN	HIGH TEMP CUT
P103-147	02/19/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	3.12	64.00	120	120	IGN	
P103-148	02/19/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	4.19	61.87	120	45	IGN	HIGH TEMP CUT
P103-149	02/19/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	4.84	117.74	300	300	IGN	
P103-150	02/22/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	4.86	120.13	300	300	IGN	
P103-151	02/22/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #1	LeRC #1	J-2	4.87	118.00	300	300	IGN	
P103-152	03/02/88	HOT FIRE CHECKOUT	LeRC #2	LeRC #2	J-2	8.00	100.00	2	2	IGN	
P103-153	03/02/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	7.98	105.40	10	10	IGN	
P103-154	03/03/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	8.14	102.67	120	19	IGN	HIGH TEMP CUT
P103-155	03/03/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	8.35	127.02	120	17	IGN	HIGH TEMP CUT
P103-156	03/03/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	8.04	75.16	120	18	IGN	HIGH TEMP CUT
P103-157	03/03/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	8.40	47.60	120	21	IGN	HIGH TEMP CUT
P103-158	03/03/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	6.00	112.27	120	24	IGN	HIGH TEMP CUT
P103-159	03/03/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	7.03	106.98	120	17	IGN	HIGH TEMP CUT
P103-160	03/03/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	4.90	117.58	120	120	IGN	
P103-161	03/03/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	3.82	82.34	120	120	IGN	
P103-162	03/03/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	3.18	63.64	120	120	IGN	
P103-163	03/03/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	4.04	90.48	120	120	IGN	
P103-164	03/03/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	5.88	140.38	120	120	IGN	
P103-165	03/03/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	7.00	150.00	120	0	NO IGN	GH2 VALVE DID NOT OPEN
P103-166	03/03/88	PERFORMANCE/COMPATABILITY DEMO	LeRC #2	LeRC #2	J-2	7.13	140.21	120	36	IGN	HIGH TEMP CUT

Table 4-1. GO_2/GH_2 25-lbf Thruster Test Log--8/26/88 (Sheet 3 of 3)

TEST NUMBER	DATE	TEST OBJECTIVE	HARDWARE CONFIGURATION INJECTOR	CHAMBER	IGNITER	TEST CONDITIONS AND STEADY STATE MIXTURE RATIO (O/F)	Pcns (psia)	DURATION SCHEDULE (sec)	DURATION ACTUAL (sec)	TEST RESULTS	REMARKS
P103-167	03/07/88	HOT FIRE CHECKOUT	LHF, 15% BLC	LeRC#1	J-2	8.00	100.00	2	2	IGN	LOW HEAT FLUX INJ., 15% BLC
P103-168	03/07/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 15% BLC	LeRC#1	J-2	8.10	97.49	10	10	IGN	LOW HEAT FLUX INJ., 15% BLC
P103-169	03/07/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 15% BLC	LeRC#1	J-2	8.00	100.46	120	120	IGN	LOW HEAT FLUX INJ., 15% BLC
P103-170	03/07/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 15% BLC	LeRC#1	J-2	8.34	123.94	120	97	IGN	LOW HEAT FLUX INJ., 15% BLC, ACCUM PRESS LOW CU
P103-171	03/07/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 15% BLC	LeRC#1	J-2	8.09	72.54	120	120	IGN	LOW HEAT FLUX INJ., 15% BLC
P103-172	03/07/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 15% BLC	LeRC#1	J-2	8.49	46.92	120	34	IGN	LOW HEAT FLUX INJ., 15% BLC, HIGH TEMP CUT T603
P103-173	03/07/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 15% BLC	LeRC#1	J-2	5.89	106.52	120	120	IGN	LOW HEAT FLUX INJ., 15% BLC
P103-174	03/07/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 15% BLC	LeRC#1	J-2	6.99	103.43	120	120	IGN	LOW HEAT FLUX INJ., 15% BLC
P103-175	03/07/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 15% BLC	LeRC#1	J-2	4.83	111.70	120	120	IGN	LOW HEAT FLUX INJ., 15% BLC
P103-176	03/07/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 15% BLC	LeRC#1	J-2	3.92	82.73	120	120	IGN	LOW HEAT FLUX INJ., 15% BLC
P103-177	03/07/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 15% BLC	LeRC#1	J-2	3.21	62.63	120	120	IGN	LOW HEAT FLUX INJ., 15% BLC
P103-178	03/07/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 15% BLC	LeRC#1	J-2	4.06	88.00	120	120	IGN	LOW HEAT FLUX INJ., 15% BLC
P103-179	03/07/88	PERFORMANCE/COMPATABILITY DEMO	LHF, 15% BLC	LeRC#1	J-2	5.86	134.10	120	82	IGN	LOW HEAT FLUX INJ., 15% BLC, DSU CUT
P103-180	03/08/88	INTERCHANGEABILITY OF HARDWARE	PROTOTYPE	LeRC#2	J-2	8.06	102.42	120	35	IGN	PROTOTYPE INJ., HIGH TEMP CUT T6034
P103-181	03/08/88	INTERCHANGEABILITY OF HARDWARE	PROTOTYPE	LeRC#2	J-2	5.96	109.49	120	120	IGN	PROTOTYPE INJ.
P103-182	03/08/88	INTERCHANGEABILITY OF HARDWARE	PROTOTYPE	LeRC#2	J-2	8.00	100.00	120	0	NO IGN	PROTOTYPE INJ.
P103-183	03/08/88	INTERCHANGEABILITY OF HARDWARE	PROTOTYPE	LeRC#2	J-2	8.00	100.00	120	0	NO IGN	PROTOTYPE INJ., VAC CABLE ON SPARK LOOSE
P103-184	03/10/88	INTERCHANGEABILITY OF HARDWARE	LeRC#2-135	LeRC#2	J-2	8.15	103.58	120	42	IGN	HIGH TEMP CUT ON T6034
P103-185	03/10/88	INTERCHANGEABILITY OF HARDWARE	LeRC#2-135	LeRC#2	J-2	8.06	103.42	120	38	IGN	HIGH TEMP CUT ON T6034
P103-186	03/10/88	INTERCHANGEABILITY OF HARDWARE	LeRC#2-135	LeRC#2	J-2	6.02	113.56	120	120	IGN	
P103-187	03/10/88	INTERCHANGEABILITY OF HARDWARE	LeRC#2-135	LeRC#2	J-2	4.84	117.32	120	120	IGN	
P103-188	03/10/88	INTERCHANGEABILITY OF HARDWARE	LeRC#2-135	LeRC#2	J-2	6.98	107.72	120	36	IGN	HIGH TEMP CUT ON T6034

To evaluate the 25-lbf thruster tested during the program, 104 tests were conducted. Three tests were inert blowdown tests conducted to verify the integrity and sequence control of the thruster integration with the test bed and test facility. The thruster did not ignite in 24 of the hot-fire attempts, as subsequently discussed; 77 tests were successfully performed to produce useful hot-fire data.

Fourteen of the nonignition tests were from 18 attempts to fire the thruster using the Simmonds Precision exciter and pressurized cable. Specific reasons for the Simmonds Precision malfunction were not successfully delineated because of time and funding constraints. Pressure loss (introducing vacuum) in the cable pressurizing sheath was a prime suspect. Later in the program it was discovered that the facility-supplied voltage (at the test bed) was below specifications on occasion. The low voltage and/or presence of a vacuum in the cable pressurizing sheath could preclude thruster ignitions.

During the remainder of the program, 10 other tests were attempted that did not produce combustion. Five of these were caused by propellant valves failing to open as a result of excessive inlet pressures supplied to the valve or by insufficient voltage supplied to open the valve. Five were caused by ignition cable sheath pressure leaks introducing vacuum around the cable.

After the unsuccessful attempts to fire the LeRC 1 thruster and Simmonds Precision igniter, this equipment was removed from the facility and replaced with the prototype thruster and J-2 exciter. After a series of five successful ignition-only prototype thruster tests, the LeRC 1 unit was reinstalled using the J-2 exciter, and no further ignition problems were encountered during the remainder of the testing. Table 4-2 summarizes the program testing. The discussion, analysis, and performance data shown herein present results obtained from the 77 successful performance, operation, and compatibility tests.

Table 4-2. Summary of Tests Performed

Test Performed/Failures	Number of Tests
Blowdown, facility	3
Simmonds Precision exciter nonignition	14
Valve overpressure/low voltage	5
Vacuum leak, nonignition	5
Performance/operation/compatibility	77
Total tests and attempts	104

Four injectors and three thrust chambers were tested during the program. The injectors consisted of the two LeRC units produced during the program, the Rocketdyne prototype unit, and an advanced version, called the "low-heat-flux injector," (LHF) produced by Rocketdyne. The low-heat-flux injector was configured for 0%, 15%, and 40% levels of boundary layer cooling (BLC).

The thrust chambers tested consisted of the Rocketdyne prototype and the two thrust chambers produced during the program. Six combinations of injectors and thrust chambers were tested, as summarized in Table 4-3. To determine if the measured uneven circumferential temperature distribution (discussed later) was caused by injector or thrust chamber effects, five tests were conducted with the LeRC 2 injector rotated 135 deg from normal position. Tests were conducted with the prototype injector assembled to the LeRC 2 thrust chamber to verify performance of the injector in the new thruster and to anchor the test results to previous prototype data. Also, measured thrust chamber circumferential and axial temperature distributions were compared to determine variations from prototype to LeRC designs.

4.1 PERFORMANCE DATA

The original data were computer printouts generated from the FM digital tapes recorded during the hot-fire testing. These data were analyzed to obtain basic thruster performance parameters and compared to predicted results and requirements.

Table 4-3. Hot-Fire Test Injector and Thrust Chamber Combinations

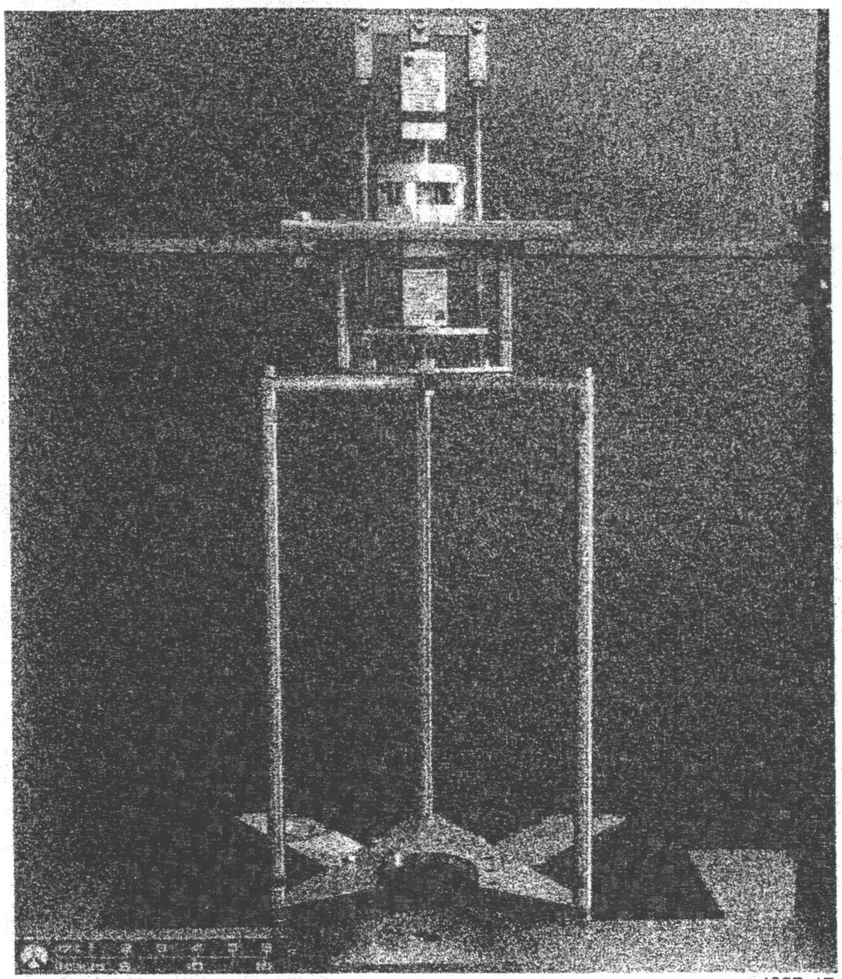
Injector	Thrust Chamber	Number of Hot Firing Tests	Accumulated Test Time (s)
LeRC 1	LeRC 1	31	2,001
LeRC 2	LeRC 2	14	764
LeRC 2[a]	LeRC 2	5	356
Prototype	Prototype	5	5
Prototype	LeRC 2	2	155
Low-heat-flux[b]	LeRC 1	20	1,374
	Total	77	4,655

[a]Tested with the injector rotated 135 deg from normal position.

[b]0%, 15%, and 40% BLC configurations were tested.

The gaseous oxidizer and gaseous fuel flow rates were measured using sonic venturis installed in the inlet tubing on the propulsion test bed. These venturis were calibrated against a standard traceable to the National Bureau of Standards. The flow rates were calculated using the calibration data and the measured venturi inlet pressures and temperatures. The pressures and temperatures were measured using accepted practices and equipment that are not elaborated upon herein.

The thrust was measured by a load cell system (Figure 4-3) designed and fabricated by Rocketdyne specifically for the 25-lbf thruster thrust measurement as part of the Freedom Station Propulsion Test Bed contract. The installed system, shown in Figure 4-2, uses a measuring load cell in series with a calibrating cell and a ram for in-place thrust calibration. The thrust system calibrations showed very linear, repeatable results, consistent with the historical data on similar systems in use.



4957-17

Figure 4-3. Load Cell System for Thrust Measurement
(15576-2/19/87-S1A*)

4.1.1 Performance Prediction

The prototype thruster data were used as a baseline for combustion performance (C*) to predict the performance of the LeRC thruster. The JANNAF performance prediction codes (Reference 2) and a Rocketdyne laminar and turbulent boundary layer analysis code were used to complete the performance modeling.

The results of the specific impulse modeling predictions, based on the prototype thruster measured C* performance over the mixture ratio range, are shown in Figure 4-4. The JANNAF prediction indicates a specific impulse of

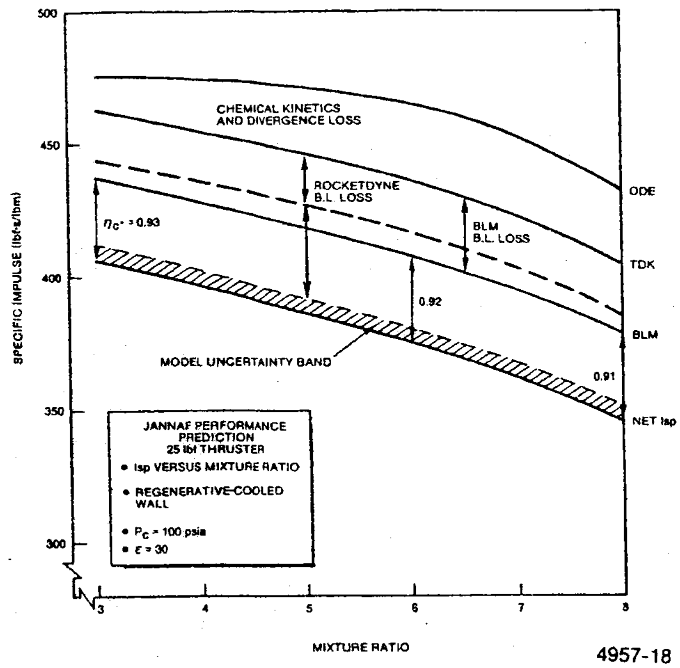


Figure 4-4. Specific Impulse Performance

406 s and 346 s at mixture ratios of 3 and 8, respectively using the measured C* efficiency. The Rocketdyne boundary layer model predicts 7 s greater specific impulse over the mixture ratio range. A predicted performance curve intersecting 346 s at MR = 8 is used in all subsequent performance graphs as a reference. The corresponding thrust coefficient and theoretical C* (JANNAF) predictions are shown in Figure 4-5.

The C* efficiency calculated from test data decreased from 93% at a mixture ratio of 3 to 91% at a mixture ratio of 8. The reasons for this characteristic are unknown but an examination of the injector design can perhaps show a reason for the decrease. The combustion efficiency produced by a coaxial injector is governed significantly by the mixing uniformity of the oxidizer and fuel within the injector elements. The mixing parameters and C* efficiency are improved as the ratio of the fuel to oxidizer injection velocity is increased. As the operating mixture ratio of the thruster is increased, the fuel flow is reduced, and this ratio is decreased. This probably results in the reduced mixing and combustion (C*) efficiency observed in testing.

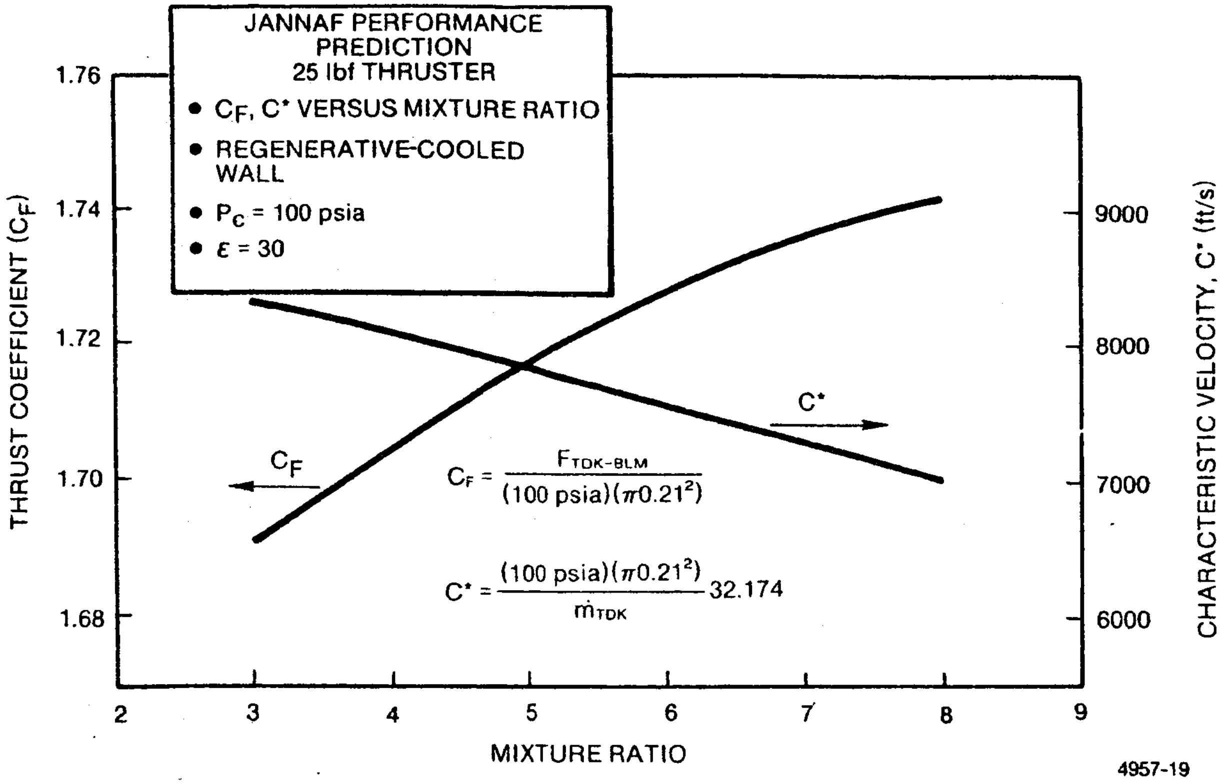


Figure 4-5. Thrust Coefficient and C* Versus Mixture Ratio

The effect of combustion chamber pressure on predicted specific impulse is shown in Figure 4-6. The specific impulse decreases by 16 s for a decrease in chamber pressure from 100 to 50 psia.

To complete the performance projection, the specific impulse calculations were expanded to show the expected results if the nozzle area ratio were to be increased from the 30:1 value used. Figure 4-7 shows the results in terms of vacuum specific impulse as a function of expansion area ratio. The 350 s value (MR = 8) obtained could be increased by 25 s to 375 s by raising the expansion area ratio to 200.

Table 1-2. 25 lbf Thruster Test Summary and Background Test Experience

INJECTOR	NOZZLE	# OF TESTS	DURATION (sec)	Pc (psia)	MR	RESULTS
PROTOTYPE	PROTOTYPE	121 10,451 PULSES	87399	45 - 106.8	3.1 - 8.1	LIFE/PERFORMANCE/PULSING DEMONSTRATION
PROTOTYPE	LeRC 1	22	135	99.3 - 114.4	6.0 - 8.0	PERFORMANCE VERIFICATION WITH NEW NOZZLE
PROTOTYPE	LeRC 2	4	155	103.7 - 111.2	6.0 - 8.1	PERFORMANCE VERIFICATION WITH NEW NOZZLE
LeRC 1	LeRC 1	26	1866	52.0 - 147.0	3.1 - 8.3	PERFORMANCE/OPERATION VERIFIED ON NEW ASSEMBLY
LeRC 2	LeRC 2	20	1324	48.6 - 142.3	3.2 - 8.4	PERFORMANCE/OPERATION VERIFIED ON NEW ASSEMBLY
LHF	LeRC 1	23	1376	46.8 - 136.0	3.2 - 8.5	Is vs %BLC ESTABLISHED LOW SKIN TEMP/LONG LIFE PREDICTED
4 INJECTORS	3 NOZZLES	216 10,451 PULSES	92142 (25.6 HRS)	45.0 - 147.0	3.1 - 8.5	CAPABILITY DEMONSTRATED

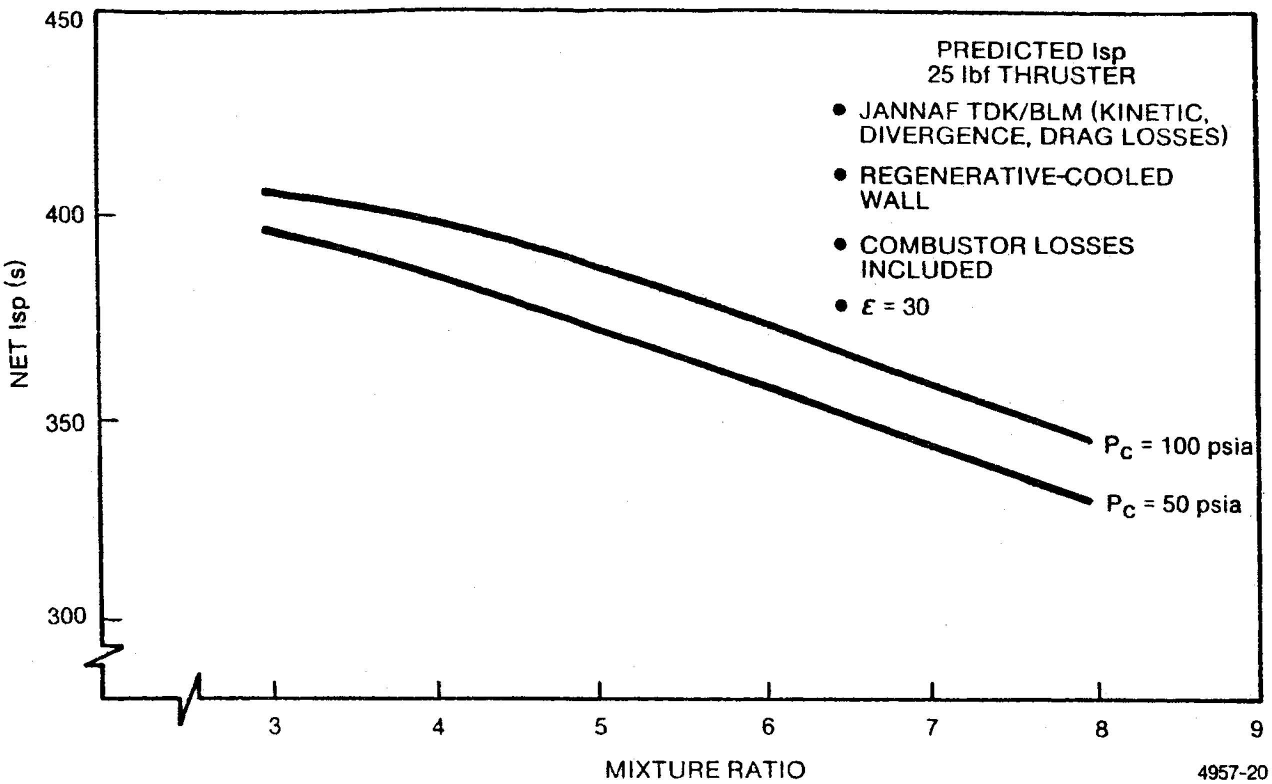


Figure 4-6. Effect of Chamber Pressure on Predicted Specific Impulse

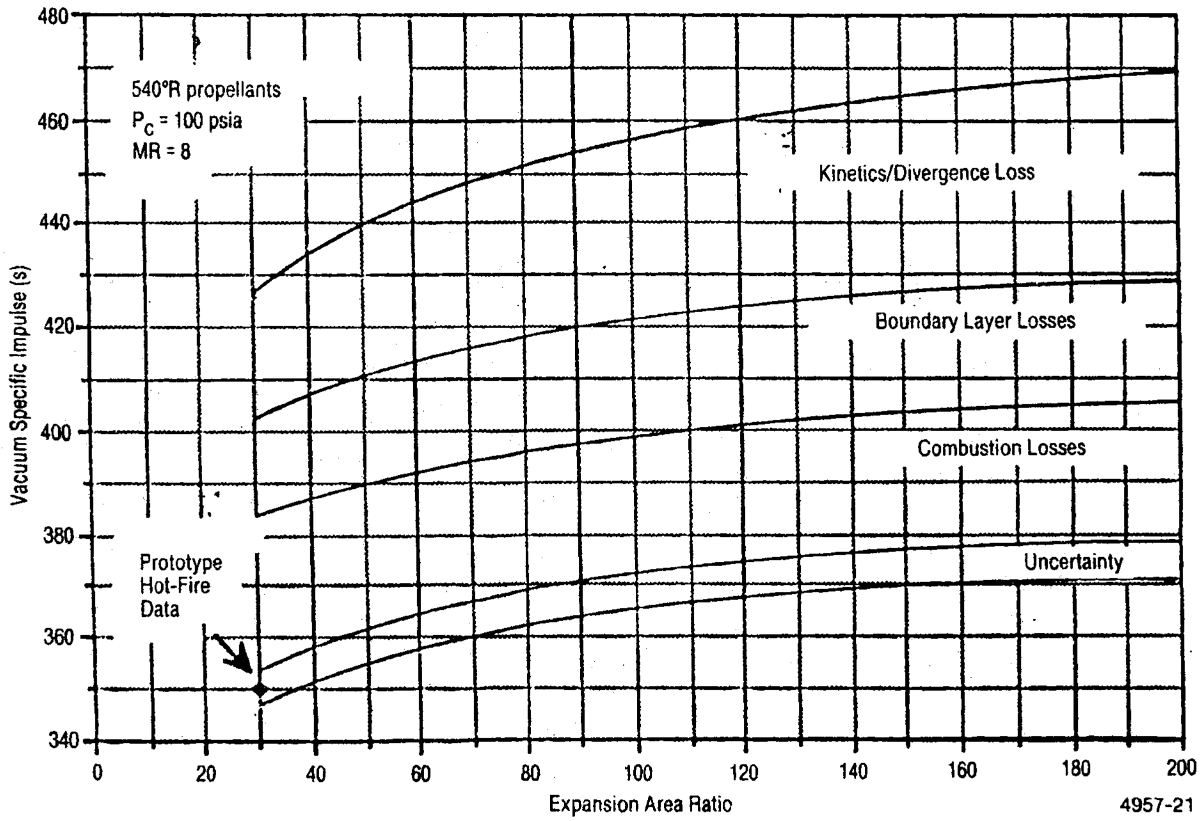


Figure 4-7. 25 lbf O_2/H_2 Thruster Performance Projection

4.1.2 Data Analysis

Table 4-4 summarizes for each test the hardware configuration, test duration, data slice time, measured thrust, fluid flows, and the calculated performance parameters. The flow rates, thrust, injector, and nozzle pressures were calculated from the original test data printouts. The table also tabulates the calculated parameters of specific impulse (I_{sp}), characteristic velocity (C*), thrust coefficient (C_F), nozzle stagnation pressure, and mixture ratio.

The specific impulse was calculated as follows:

$$I_{sp} = \frac{F}{W_O + W_F} \tag{1}$$

The thrust coefficient was calculated as follows:

$$C_F = \frac{F}{0.995\ P_{cns}\ At}$$; (0.995 = nozzle throat flow discharge coefficient)

The C* value was calculated from the hot-fire data as follows:

$$C^* = \frac{P_{cns}\ A_t g}{W_o + W_f} \tag{2}$$

where

W_O = oxidizer flow rate (lb/s) =
W_F = fuel flow rate (lb/s) =
F = vacuum thrust (lb),
= $F_{measured} + A_e P_v$
A_t = nozzle throat area (in^2)
P_c = measured chamber pressure (psia)
I_{sp} = thruster specific impulse (s) (vacuum)
C_F = nozzle thrust coefficient (dimensionless) (vacuum)

Table 4-4. Thruster Operating Parameters
(Sheet 1 of 2)

					THRUSTER OPERATING PARAMETERS											
TEST NO.	INJECTOR	CHAMBER	TEST DURATION (sec)	DATA SLICE TIME (sec)	Pcns (psia)	MR	Thrust (lbf)	Oxid Flow (lb/sec)	Fuel Flow (lb/sec)	PO2 inj (psia)	PH2 inlet (psia)	PH2 chamber (psia)	PH2 inj (psia)	C* (ft/sec)	Cf (vac)	Is (sec)
P103-153	LeRC#2	LeRC#2	10	9.6	105.40	7.98	25.87	0.0662	0.0083	146.02	289.77	169.99	119.78	6343	1.762	347.37
P103-154	LeRC#2	LeRC#2	19	19.1	102.67	8.14	24.99	0.0645	0.0079	142.96	282.56	163.52	119.04	6369	1.744	345.31
P103-155	LeRC#2	LeRC#2	17	17.3	127.02	8.35	31.17	0.0799	0.0096	176.62	339.05	192.85	146.20	6377	1.759	348.62
P103-156	LeRC#2	LeRC#2	18	18.8	75.16	8.04	18.31	0.0482	0.0060	106.76	214.88	127.15	87.73	6227	1.746	337.88
P103-157	LeRC#2	LeRC#2	21	20.0	47.60	8.40	11.51	0.0323	0.0039	70.91	138.95	138.95	55.29	5904	1.734	318.11
P103-158	LeRC#2	LeRC#2	24	24.6	112.27	6.00	27.37	0.0622	0.0104	148.56	365.82	227.06	138.76	6951	1.746	377.27
P103-159	LeRC#2	LeRC#2	17	17.1	106.98	7.03	26.20	0.0635	0.0090	145.74	319.52	191.99	127.53	6627	1.756	361.58
P103-160	LeRC#2	LeRC#2	120	30.1	117.58	4.90	28.79	0.0606	0.0124	150.71	423.40	272.56	150.84	7240	1.754	394.60
P103-161	LeRC#2	LeRC#2	120	30.0	82.34	3.82	20.06	0.0395	0.0104	103.73	343.26	230.01	113.25	7416	1.744	401.99
P103-162	LeRC#2	LeRC#2	120	30.0	63.64	3.18	15.45	0.0290	0.0091	79.38	299.06	206.05	93.01	7489	1.738	404.64
P103-163	LeRC#2	LeRC#2	120	30.1	90.48	4.04	22.14	0.0445	0.0110	114.35	363.80	241.43	122.37	7325	1.752	398.93
P103-164	LeRC#2	LeRC#2	120	30.0	140.38	5.88	34.90	0.0769	0.0131	184.71	445.62	270.70	174.92	7007	1.780	387.66
P103-166	LeRC#2	LeRC#2	36	30.0	140.21	7.13	34.88	0.0830	0.0117	191.37	405.39	236.05	169.34	6657	1.781	368.53
P103-180	PROTOTYPE	LeRC#2	35	30.0	102.42	8.06	25.47	0.0648	0.0080	139.78	283.89	160.20	123.69	6322	1.781	349.87
P103-181	PROTOTYPE	LeRC#2	120	30.0	109.49	5.96	27.35	0.0618	0.0104	143.41	353.86	216.27	137.59	6816	1.789	378.87
P103-184	LeRC#2-135	LeRC#2	42	30.0	103..58	8.15	24.89	0.0650	0.0080	144.98	278.85	155.73	123.12	6376	1.721	341.01
P103-185	LeRC#2-135	LeRC#2	38	30.0	103.42	8.06	24.64	0.0649	0.0081	145.27	280.36	157.11	123.25	6370	1.706	337.80
P103-186	LeRC#2-135	LeRC#2	120	30.0	113.56	6.02	27.30	0.0625	0.0104	150.16	359.42	215.94	143.48	6998	1.721	374.34
P103-187	LeRC#2-135	LeRC#2	120	30.0	117.32	4.84	28.19	0.0594	0.0123	150.16	417.34	262.12	155.22	7367	1.720	393.38
P103-188	LeRC#2-135	LeRC#2	120	30.0	107.72	6.98	25.48	0.0640	0.0092	147.72	318.67	186.13	132.54	6615	1.694	348.23

Table 4-4. Thruster Operating Parameters
(Sheet 2 of 2)

			THRUSTER OPERATING PARAMETERS													
TEST NO.	INJECTOR	CHAMBER	TEST DURATION (sec)	DATA SLICE TIME (sec)	Pcns (psia)	MR	Thrust (lbf)	Oxid Flow (lb/sec)	Fuel Flow (lb/sec)	PO2 inj (psia)	PH2 inlet (psia)	PH2 chamber (psia)	PH2 inj (psia)	C* (ft/sec)	Cf (vac)	Is (sec)
P103-057	LeRC#1	LeRC#1	10	9.2	102.37	8.05	25.17	0.0638	0.0079	143.80	237.10	121.80	115.30	6396	1.767	351.25
P103-071	LeRC#1	LeRC#1	10	9.6	102.30	7.71	25.00	0.0647	0.0084	111.26	243.52	126.45	117.07	6277	1.754	342.18
P103-072	LeRC#1	LeRC#1	30	29.5	99.87	7.61	24.22	0.0609	0.0080	108.40	252.19	132.34	119.85	6511	1.737	351.39
P103-073	LeRC#1	LeRC#1	30	29.8	111.01	6.02	26.85	0.0619	0.0103	115.55	313.97	175.33	138.64	6911	1.732	372.03
P103-074	LeRC#1	LeRC#1	52	29.7	112.65	6.08	26.63	0.0616	0.0101	149.67	312.62	172.62	140.00	7061	1.693	371.43
P103-127	LeRC#1	LeRC#1	10	9.7	105.56	8.18	26.34	0.0654	0.0080	145.78	241.55	118.95	122.60	6444	1.791	358.78
P103-129	LeRC#1	LeRC#1	28	28.3	107.90	8.15	26.91	0.0653	0.0080	150.24	256.62	126.67	129.95	6611	1.786	366.95
P103-130	LeRC#1	LeRC#1	26	26.0	110.70	7.02	27.47	0.0634	0.0090	150.29	283.23	145.89	137.34	6870	1.777	379.35
P103-133	LeRC#1	LeRC#1	39	30.0	139.14	7.28	36.58	0.0805	0.0111	189.09	342.24	171.83	170.41	6833	1.882	399.73
P103-134	LeRC#1	LeRC#1	32	30.0	130.46	8.25	32.76	0.0796	0.0097	182.10	300.74	144.78	155.96	6570	1.798	367.21
P103-135	LeRC#1	LeRC#1	120	30.0	118.88	4.85	29.61	0.0597	0.0123	152.39	368.01	209.51	158.50	7418	1.783	411.17
P103-136	LeRC#1	LeRC#1	22	19.6	111.82	5.92	28.08	0.0606	0.0102	147.93	313.62	169.82	143.80	7090	1.800	396.59
P103-137	LeRC#1	LeRC#1	27	26.4	87.80	4.99	21.65	0.0450	0.0090	114.56	275.06	156.17	118.89	7303	1.766	400.79
P103-138	LeRC#1	LeRC#1	29	22.0	57.44	5.00	14.04	0.0301	0.0060	76.09	186.41	107.43	78.98	7131	1.751	388.16
P103-139	LeRC#1	LeRC#1	20	20.1	51.92	7.34	12.90	0.0320	0.0044	74.11	138.25	73.93	64.32	6419	1.780	355.16
P103-140	LeRC#1	LeRC#1	24	24.3	104.64	8.07	26.25	0.0643	0.0080	147.21	248.46	122.18	126.28	6509	1.797	363.49
P103-141	LeRC#1	LeRC#1	28	28.1	107.17	8.16	25.65	0.0651	0.0080	148.97	249.51	120.40	129.11	6591	1.714	351.11
P103-142	LeRC#1	LeRC#1	120	30.0	118.86	5.31	28.77	0.0610	0.0115	153.73	346.47	190.34	156.13	7365	1.733	396.74
P103-143	LeRC#1	LeRC#1	120	30.0	144.98	5.89	35.75	0.0772	0.0132	189.79	396.12	210.00	186.12	7213	1.766	395.81
P103-144	LeRC#1	LeRC#1	17	17.9	93.00	4.14	22.41	0.0449	0.0109	117.23	314.99	185.42	129.57	7494	1.727	402.30
P103-145	LeRC#1	LeRC#1	120	30.1	93.66	4.11	22.84	0.0449	0.0109	118.17	319.37	187.60	131.77	7533	1.746	408.75
P103-146	LeRC#1	LeRC#1	17	17.3	80.13	6.97	19.65	0.0471	0.0068	110.33	208.44	108.77	99.67	6674	1.757	364.94
P103-147	LeRC#1	LeRC#1	120	30.0	64.00	3.12	15.37	0.0282	0.0091	79.16	259.19	160.36	98.83	7716	1.719	412.28
P103-148	LeRC#1	LeRC#1	45	30.0	61.87	4.19	14.81	0.0304	0.0073	79.83	217.45	127.79	89.66	7380	1.714	393.20
P103-149	LeRC#1	LeRC#1	300	29.5	117.74	4.84	29.15	0.0594	0.0123	151.16	364.48	205.56	158.92	7387	1.773	407.02
P103-150	LeRC#1	LeRC#1	300	30.0	120.13	4.86	29.91	0.0603	0.0124	160.88	364.30	210.93	153.37	7427	1.783	411.50
P103-151	LeRC#1	LeRC#1	300	30.0	118.00	4.87	29.68	0.0598	0.0123	151.89	362.61	203.34	159.27	7352	1.801	411.50

A_e = nozzle exit area (in^2) (nominal = 4.231 in^2)
P_v = ambient pressure (psi)
P_{cns} = nozzle stagnation pressure: 0.992 (P_c-0.81); (0.992 = combustion chamber flow contraction ratio correction and 0.81 = pressure loss from point of measurement through the injector face igniter port to the combustion chamber.)
g = the gravitational constant 32.2 ft/s^2.

It should be noted that the throat area, A_t, was affected by nozzle temperature, increasing as the nozzle heated during firing. Using the measured temperatures, the variation in throat area was established as a function of thruster firing time as

$$A_t = A_o(1+\varepsilon\Delta T)^2 \tag{3}$$

where

A_o = throat area at ambient temperature (0.1385 in^2)
α = NARloy-Z thermal expansion coefficient (8 x 10^{-6} in/in.°F)
ΔT = measured temperature change.

4.1.3 Thruster Performance Results

Variations in thruster performance parameters (C*, I_{sp}) as a function of run time were observed. Figures 4-8 and 4-9 show the variation in C* efficiency and specific impulse (I_{sp}) with run time for three representative tests for both LeRC 1 and the prototype thruster. This variation was attributed to improved hydrogen and oxygen mixing resulting from increased hydrogen injection temperature and velocity as the hydrogen temperature increased during the first 20 s($\pm$) of thruster firing (see Section 4.1.1).

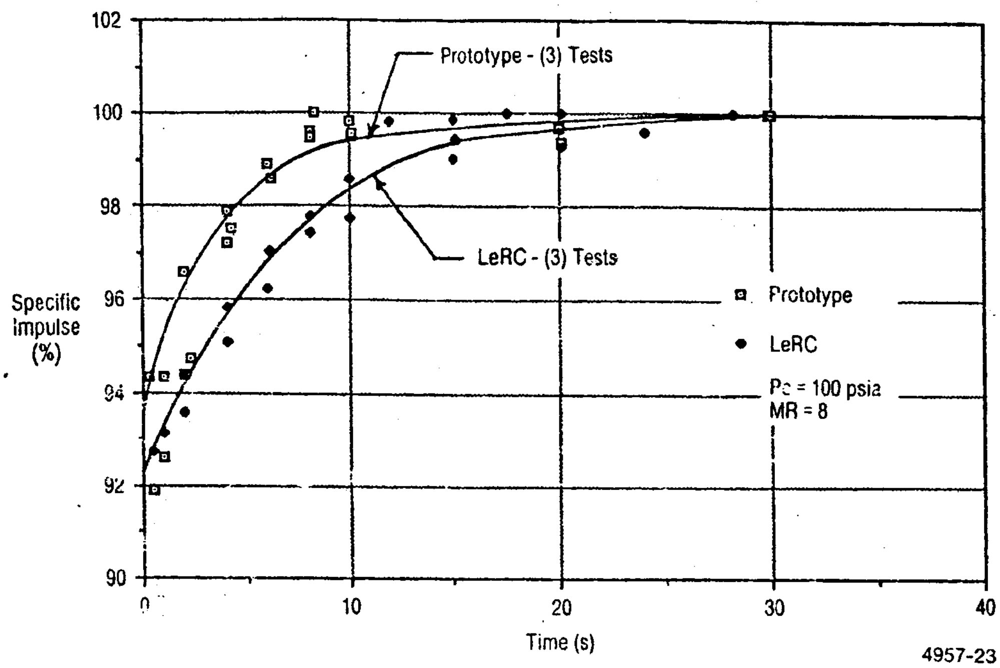


Figure 4-8. Performance Variations with Run Time (A)

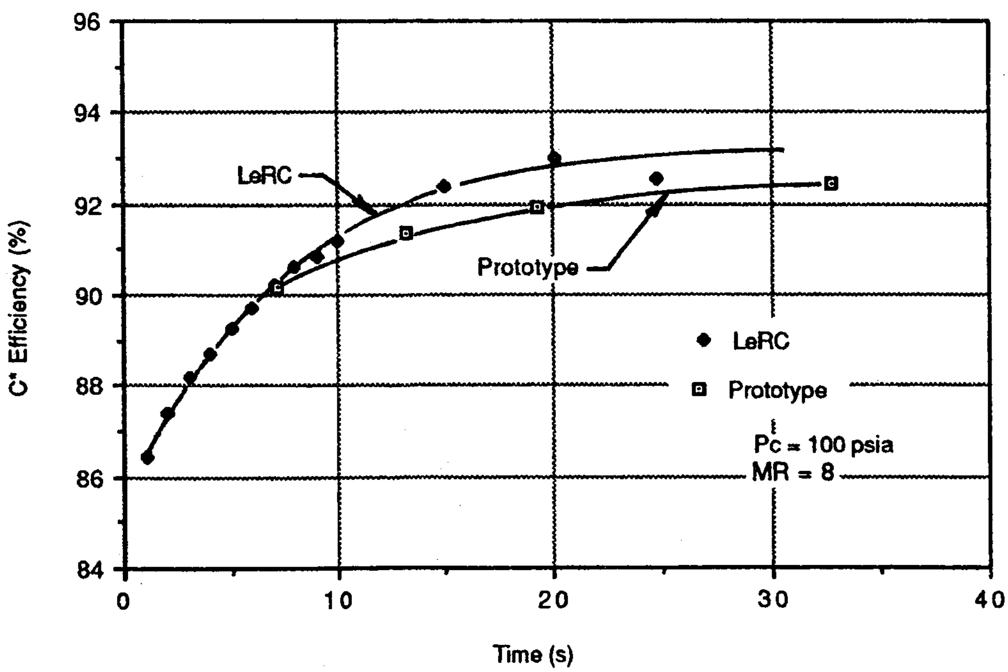


Figure 4-9. Performance Variations with Run Time (B)

The data indicate that 15 s(±) of thruster run time are required to stabilize the rise in C* and I_{sp}. Specific impulse performance data presented are shown at a time slice of 15 s (or more) to permit direct comparisons and eliminate variations caused by time dependency early in the firing.

The thruster specific impulse data are all presented as a function of mixture ratio at a time slice of approximately 20 s measured from fuel valve open signal. The predicted values, as described in Section 4.1.1, are shown for reference.

Specific impulse and thrust coefficient for the LeRC thrusters calculated from the test data are presented in Figures 4-10a and 4-10b. The data scatter evident in the figures calculated from measured thrust and flow rates is indicative of thrust measurement inconsistency. The data scatter was attributed to structural hysteris in the thruster inlet plumbing and to some variations in thrust calibration procedures. No reasons were found to cause the pressure measurements or propellant flow measurements to be suspect.

Specific impulse calculated data are presented in Figure 4-11 for the prototype thruster and for the prototype injector installed in the LeRC 2 thrust chamber. The data follow the predicted curve quite well.

The LeRC 2 thruster assembly was tested in two configurations. One series of tests (P103-184 through -188) was performed with the injector to thrust chamber orientation rotated 135 deg from nominal position. This change was made to determine if the circumferential heating effects were altered. All other tests were performed with the injector oriented normally.

An injector configuration that had been designed and fabricated by Rocketdyne to reduce the heat flux to the thruster and enhance thruster life was tested during the same period of this program. Data from tests of this low-heat-flux injector with the LeRC 1 thrust chamber are presented in Figure 4-12. Testing was performed with combustor zone BLCs of 0, 15, and 40%.

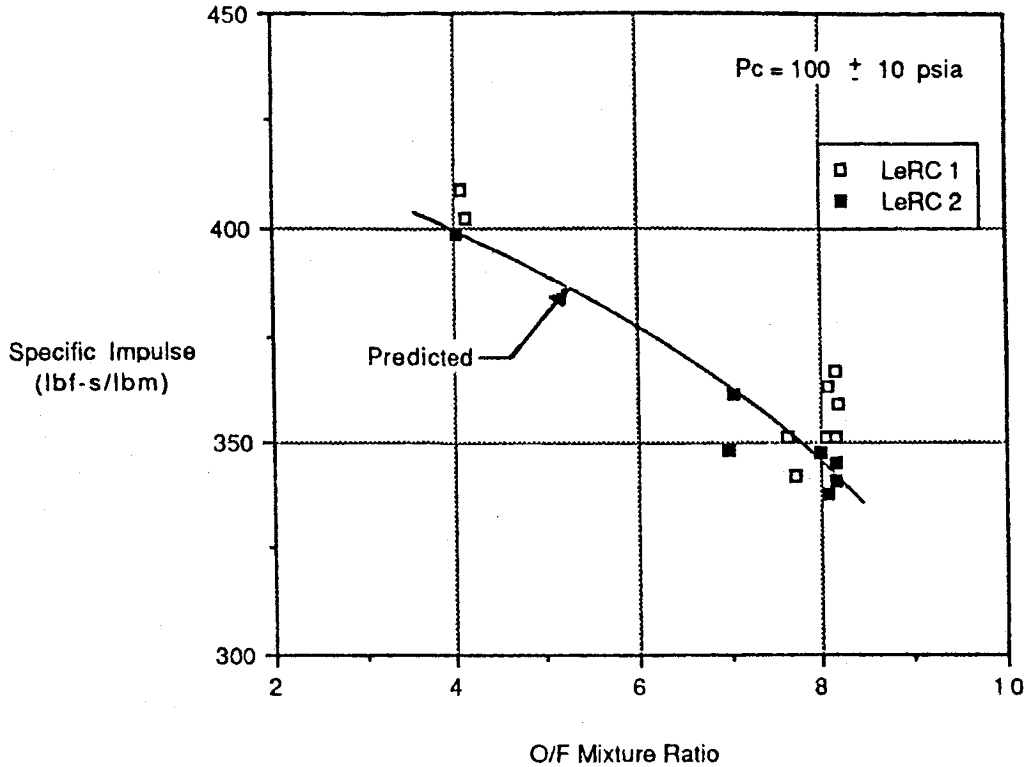


Figure 4-10a. Specific Impulse

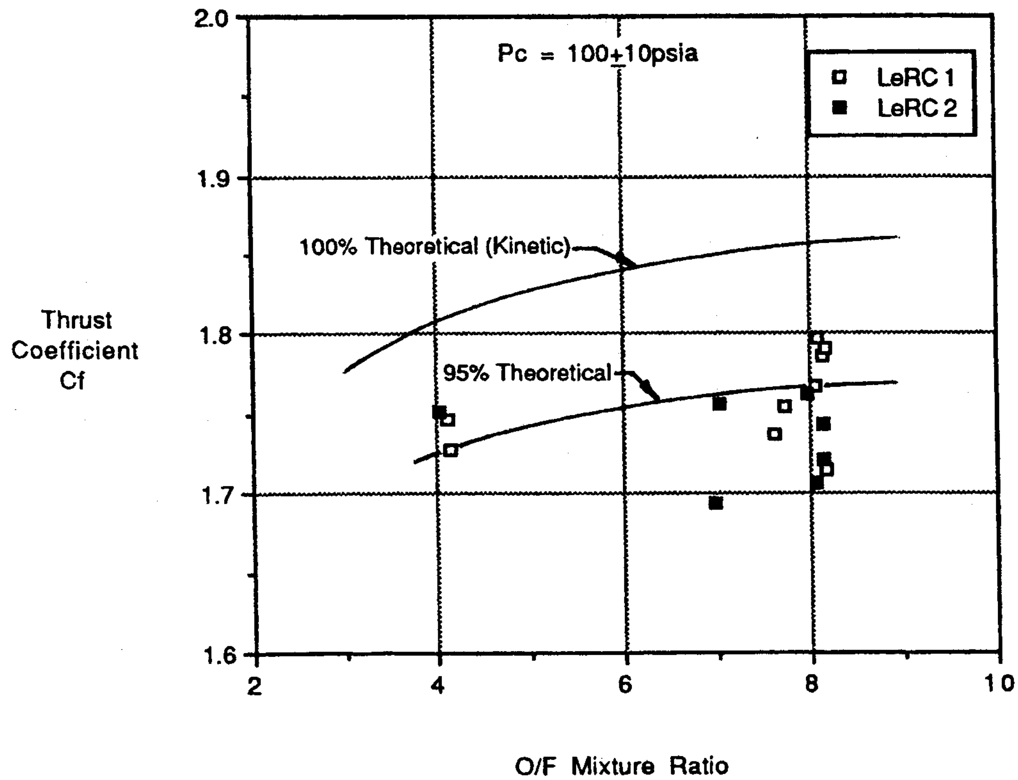


Figure 4-10b. Thrust Coefficient Variations

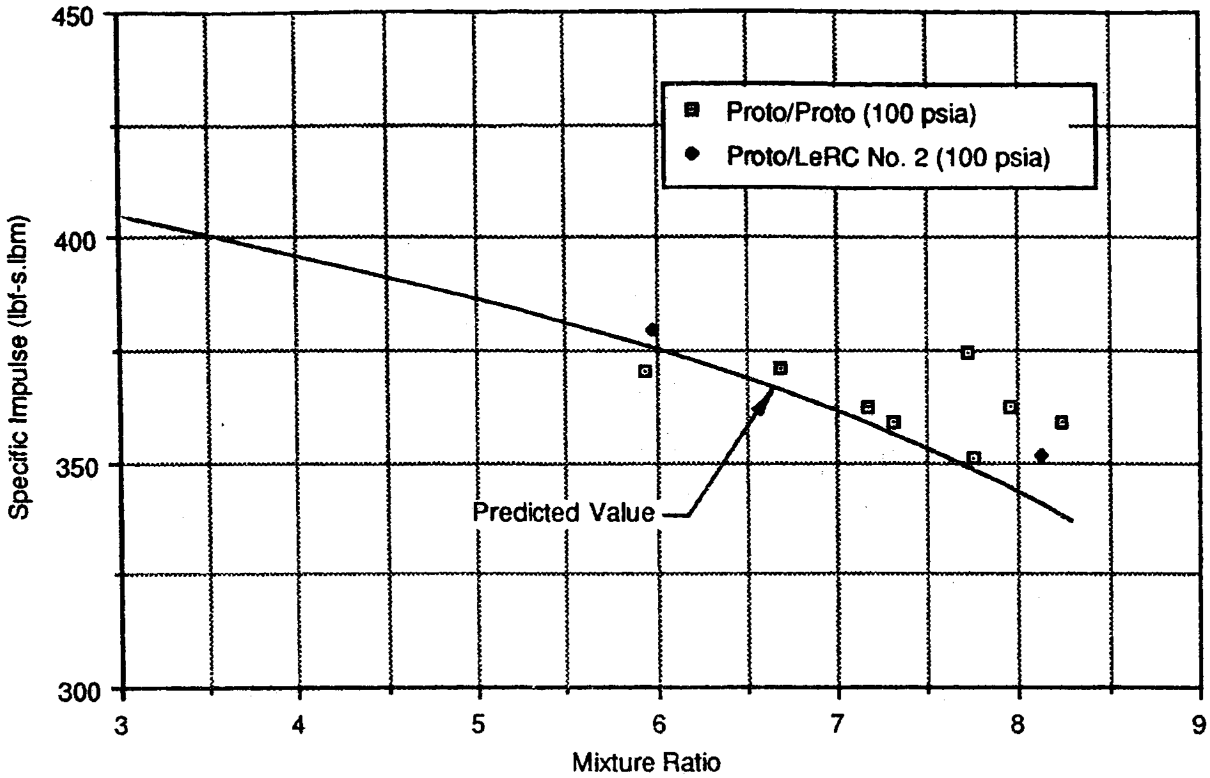


Figure 4-11. 25-lbf Thruster - Prototype Injector Performance (20 s)

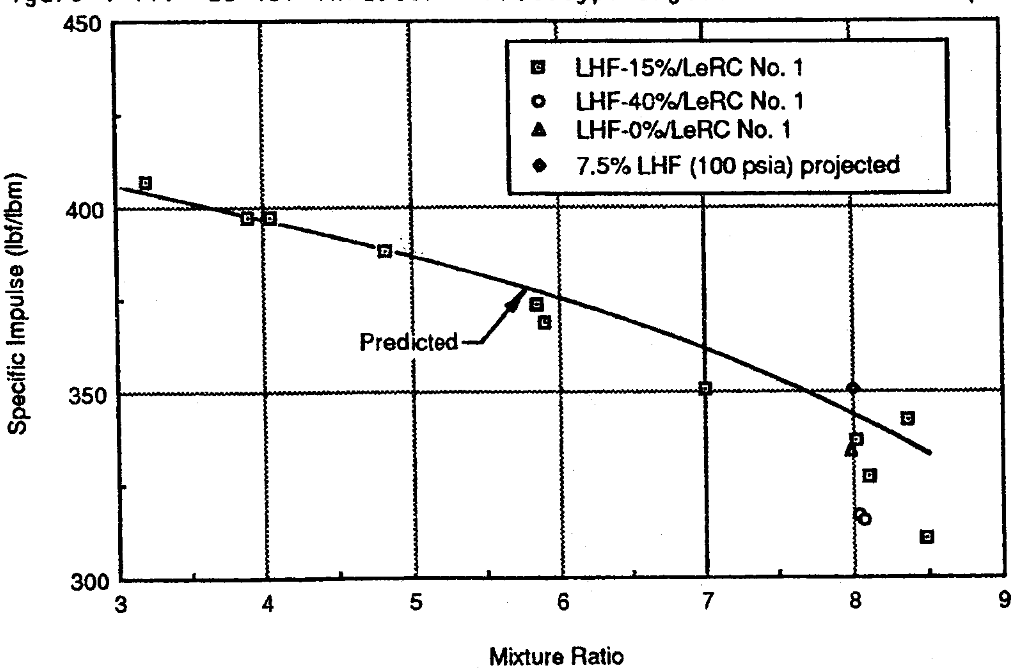


Figure 4-12. 25-lbf Thruster - Low-Heat-Flux Injector Performance (20 s)

A specific impulse at a mixture ratio of 8 at 355 s for 0% BLC, 337 s for 15% BLC, and 332 s for 40% BLC was observed. The recommended design of this low-heat-flux injector (LHF) is predicted to produce 350 s of specific impulse at a mixture ratio of 8, which is comparable to the LeRC 1 and 2, and prototype results. Significant combustion chamber heat transfer reductions were demonstrated.

4.1.3.1 Chamber Pressure Effects. The effects of chamber pressure on specific impulse are shown in Figure 4-13. The hot-firing specific impulse data followed the theoretical prediction well, but had a tendency to drop off the predicted curve at lower chamber pressures (50 psia) by approximately 5 lbf-s/lbm.

4.1.3.2 Pulsing Performance. Pulsing performance can be defined as the impulse generated by an actual thrust pulse considering thrust buildup, propellant leads, and shutdown sequencing actually obtainable as a percentage of the impulse that would be generated by the same quantity of propellant burned at the nominal thruster steady-state operation conditions. The following was used to evaluate pulsing performance obtainable with the thruster:

$$\frac{\int_{t=0}^{t=(F=0)} F\,dt}{350\ tp} \tag{4}$$

where

t_p = nominal pulse duration

The nominal pulse duration, t_p, was defined as the time between oxidizer valve opening and closing signals. A typical pulse transient is shown in Figure 4-14. An analysis was performed using the procedure outlined previously, assuming a fuel lead at start of 20 ms and 20 ms fuel lag at cutoff.

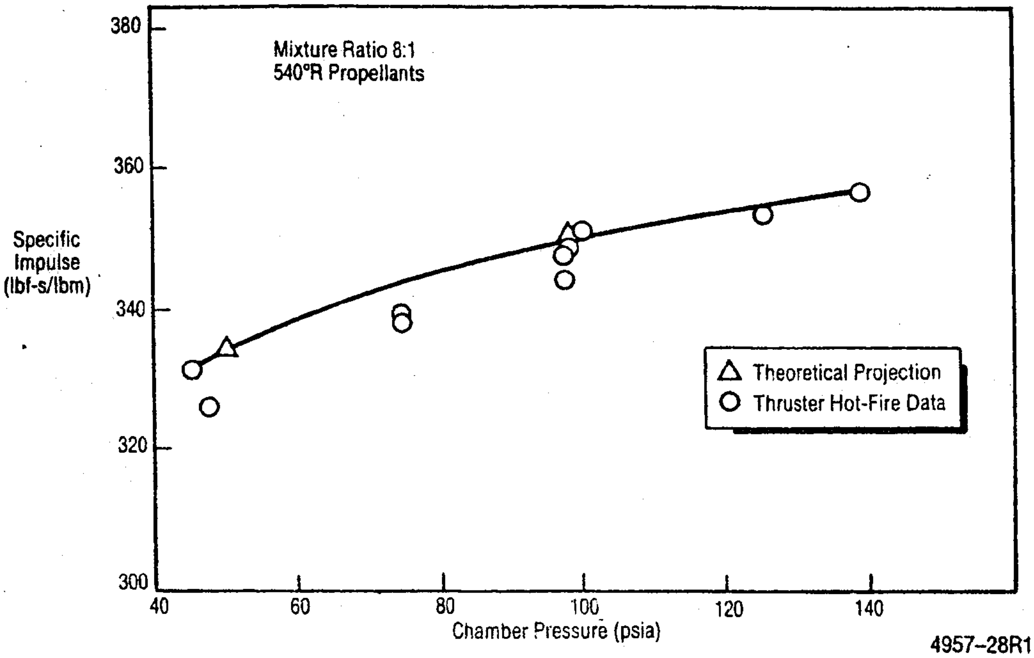


Figure 4-13. 25-lbf O_2/H_2 Thruster Performance Projection Chamber Pressure Effects

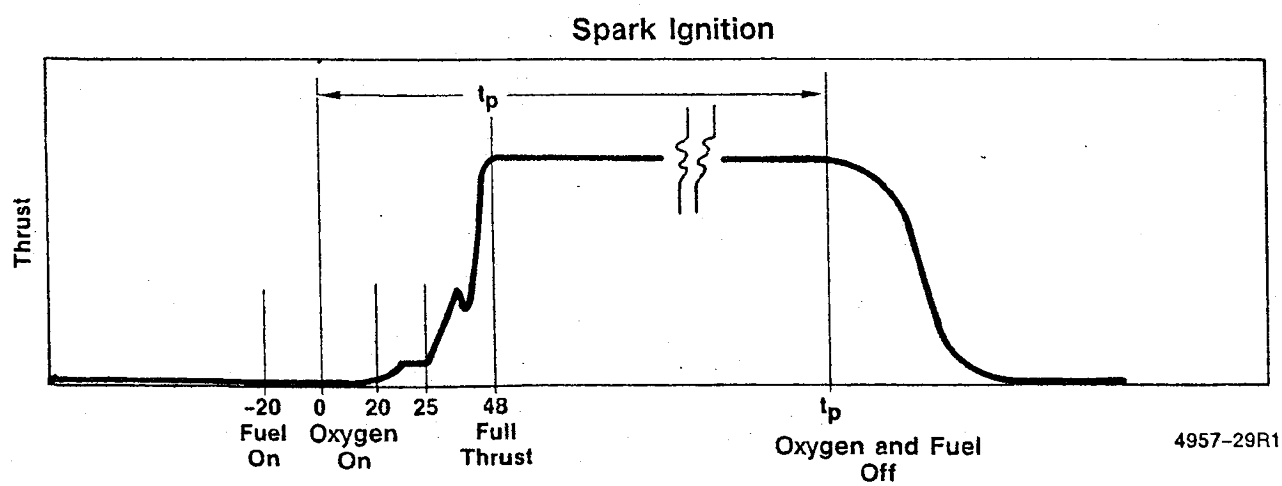


Figure 4-14. Pulse Profile

The results are shown in Figure 4-15 as percent of steady-state impulse obtainable as a function of pulse width. To prevent a drop in performance for pulse times of less than 200 ms, the fuel lead at start and lag at cutoff could be reduced to near zero. Since losses always occur in the start and cutoff, a performance of 100% cannot be achieved.

4.1.3.3 Thruster Thermal Characteristics. The thruster assembly was instrumented with thermocouples on the external surface to monitor thruster thermal behavior during the hot-fire tests. Figure 4-16 displays thermocouple placement and identification on the thruster. A significant parameter used to evaluate thrust chamber heating and injector/combustor performance is the thrust chamber hydrogen coolant temperature rise. Data from firings of the prototype thruster and injector fired with the LeRC 2 thrust chamber are shown in Figure 4-17. The hydrogen temperature rise for the prototype thruster was 900°F after 40 s of operation and the prototype/LeRC 2 combination would reach about 850°F. The time-to-temperature relationship is similar for the two thrust chambers. To reach thermal equilibrium, the thruster requires a 30- to 40-s firing time.

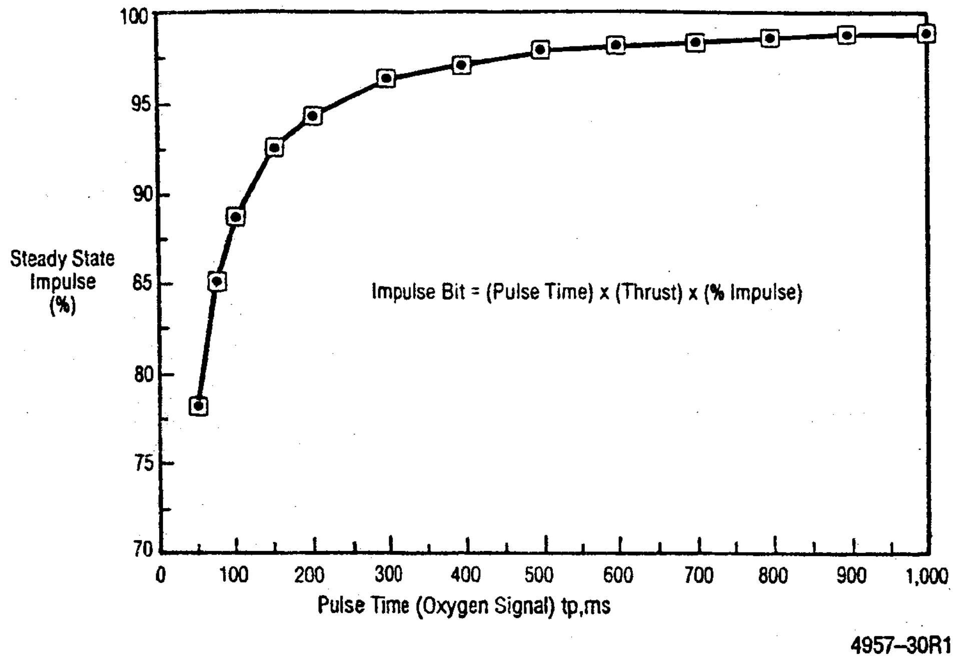
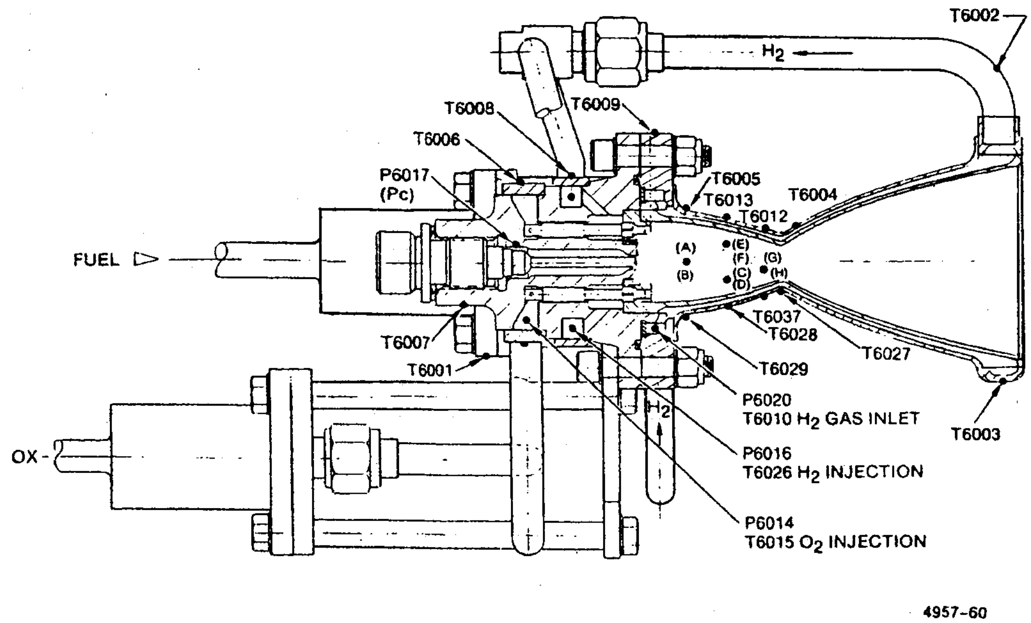


Figure 4-15. Pulsing Performance

Figure 4-18 displays the time temperature behavior of the nozzle coolant hydrogen for the prototype, LeRC 1 and 2, and low-heat-flux (LHF) injector with the LeRC 1 thrust chamber. As shown in Figure 4-18, the LeRC 1 and 2 characteristics are similar to the prototype injector/LeRC 2 thrust chamber combination.

The LHF injector produces a significantly reduced hydrogen coolant temperature rise of 670°F, which is approximately 230°F less than the prototype and 180°F less than the LeRC assembly. The LHF injector configuration would, for the LeRC combustors, represent 79% of the total heat flux produced by the LeRC injector configuration.

External hardware temperatures recorded during the hot-fire tests are displayed in Figures 4-19 and 4-20. At each axial section of the thrust chamber, the recorded temperatures were averaged to obtain the information shown.



INSTRUMENTATION

DESIGNATION	DESCRIPTION	x	θ
T6001	H2 VALVE BRACKET TEMPERATURE	-3.5	247.5
T6002	H2 RETURN TUBE TEMPERATURE	2.6	0
T6003	H2 EXIT MANIFOLD TEMPERATURE	2.6	180
T6004	NOZZLE TEMPERATURE, DOWNSTREAM OF THROAT	-0.2	0
T6005	TEMPERATURE AT UPSTREAM END OF CHAMBER	-1.0	0
T6006	INJECTOR TEMPERATURE AT O2 MANIFOLD	-3.0	0
T6007	INJECTOR TEMPERATURE AT SPARK PLUG	-3.7	0
T6008	INJECTOR TEMPERATURE AT H2 MANIFOLD	-2.4	0
T6009	THROAT CHAMBER FLANGE TEMPERATURE	-1.5	0
T6010	H2 INLET GAS TEMPERATURE	-1.5	67.5
T6012	NOZZLE TEMPERATURE UPSTREAM OF THROAT	-0.2	0
T6013	MID COMBUSTION CHAMBER TEMPERATURE	-0.7	0
T6015	O2 INJECTION TEMPERATURE	-3.0	22.5
T6026	H2 INJECTION TEMPERATURE	-2.4	22.5
T6027	CHAMBER THROAT TEMPERATURE	0	180
T6028	MID COMBUSTION CHAMBER TEMPERATURE	-0.7	180
T6029	TEMPERATURE AT UPSTREAM END OF CHAMBER	-1.0	180
T6030 (A)	TEMPERATURE AT UPSTREAM END OF CHAMBER	-1.0	90
T6031 (B)	TEMPERATURE AT UPSTREAM END OF CHAMBER	-1.0	270
T6032 (C)	MID COMBUSTION CHAMBER TEMPERATURE	-0.7	240
T6033 (D)	MID COMBUSTION CHAMBER TEMPERATURE	-0.7	120
T6034 (E)	MID COMBUSTION CHAMBER TEMPERATURE	-0.7	60
T6035 (F)	MID COMBUSTION CHAMBER TEMPERATURE	-0.7	300
T6036 (G)	NOZZLE TEMPERATURE UPSTREAM OF THROAT	-0.2	270
T6037	NOZZLE TEMPERATURE UPSTREAM OF THROAT	-0.2	180
T6038 (H)	NOZZLE TEMPERATURE UPSTREAM OF THROAT	-0.2	90
T6039	H2 RETURN TUBE EXIT TEMPERATURE	-1.5	0
P6014	O2 INJECTION PRESSURE	-3.0	22.5
P6016	H2 INJECTION PRESSURE	-2.4	22.5
P6017	CHAMBER PRESSURE	-3.7	45
P6020	H2 INLET MANIFOLD PRESSURE	-1.5	67.5

x = Measured from throat plane (+ downstream, - upstream)
θ = 0 at return tube (clockwise looking downstream)

For an expanded view of the thruster, refer to thruster assembly, Figure A-1, 7R033601 in Appendix A.

Figure 4-16. NASA-LeRC 25-lbf GO_2/GH_2 Thruster

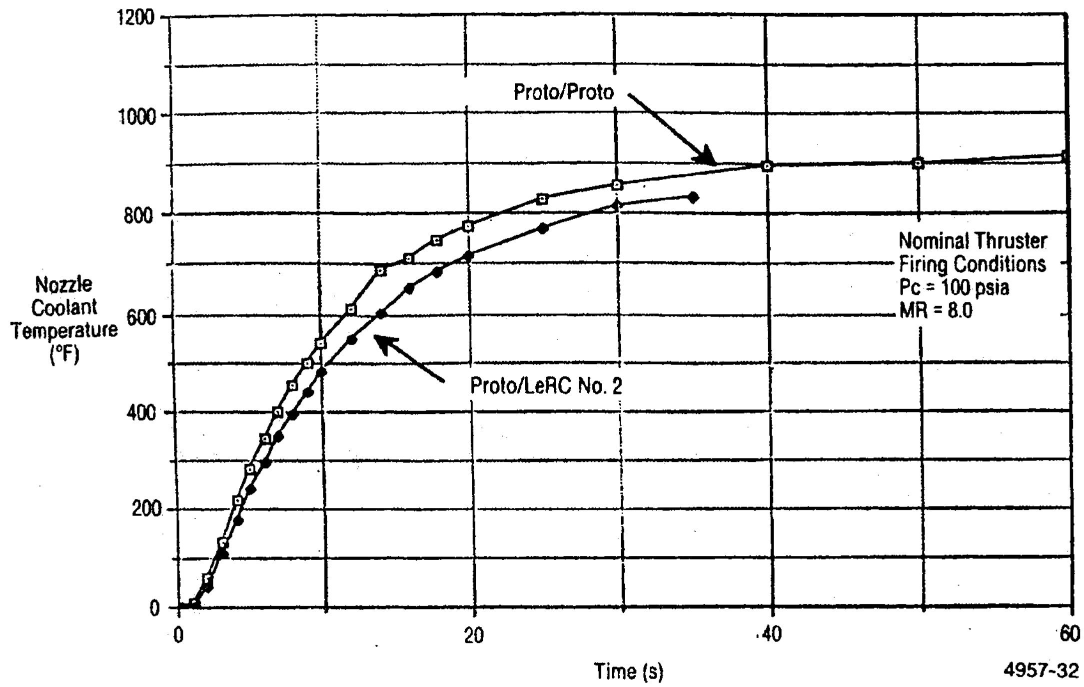


Figure 4-17. 25-lbf GO_2/GH_2 Thruster H_2 Nozzle Coolant Temperature Rise

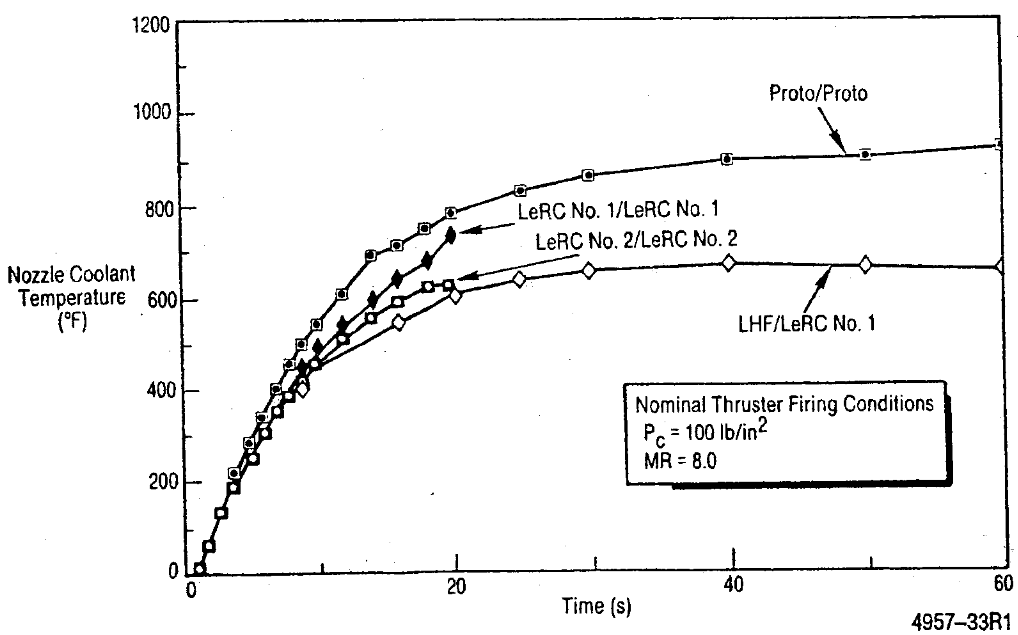


Figure 4-18. 25-lbf GO_2/GH_2 Thruster Nozzle Coolant Temperature Rise

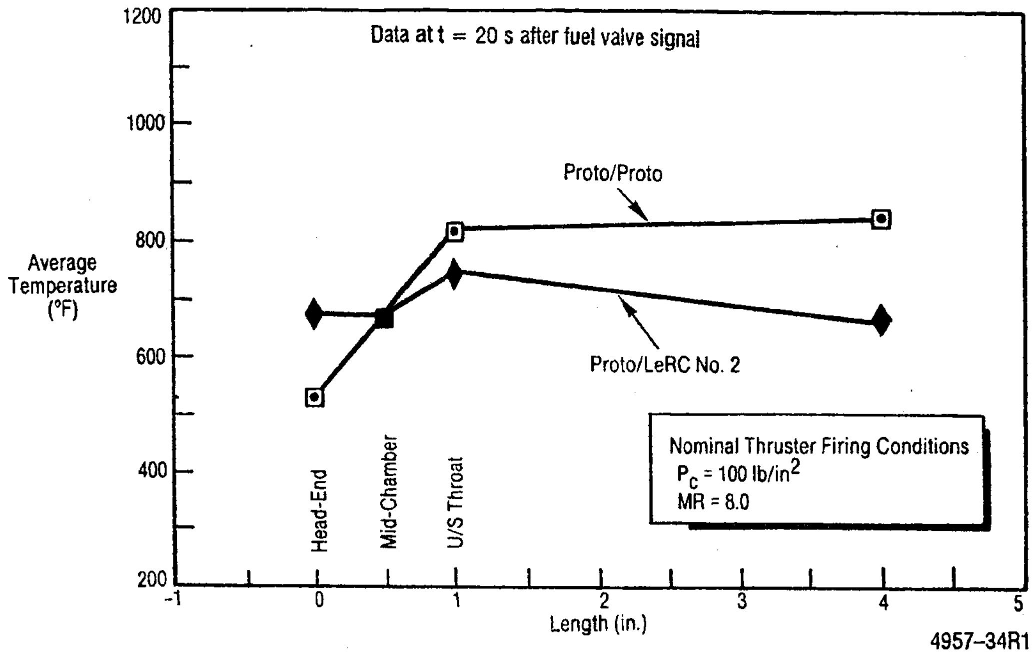


Figure 4-19. 25-lbf GO_2/GH_2 Thruster Comparison of Temperatures for Prototype and LeRC 2

A comparison of the prototype thruster data and the prototype injector/ LeRC 2 thrust chamber data (Figure 4-19) indicates the LeRC thrust chamber runs hotter toward the chamber injector end and cooler in the nozzle section than the prototype thrust chamber. This result was typical of the new injectors (Figure 4-20).

The differences in temperatures observed between LeRC 1 and 2 thrust chambers were attributed to differences in the manufacture of the units. A detailed discussion of these differences is given in Section 3.1.

The measured thrust chamber wall temperatures were significantly reduced when the LHF injector was used. The average temperature in the thrust chamber combustion zone was reduced by approximately 300°F.

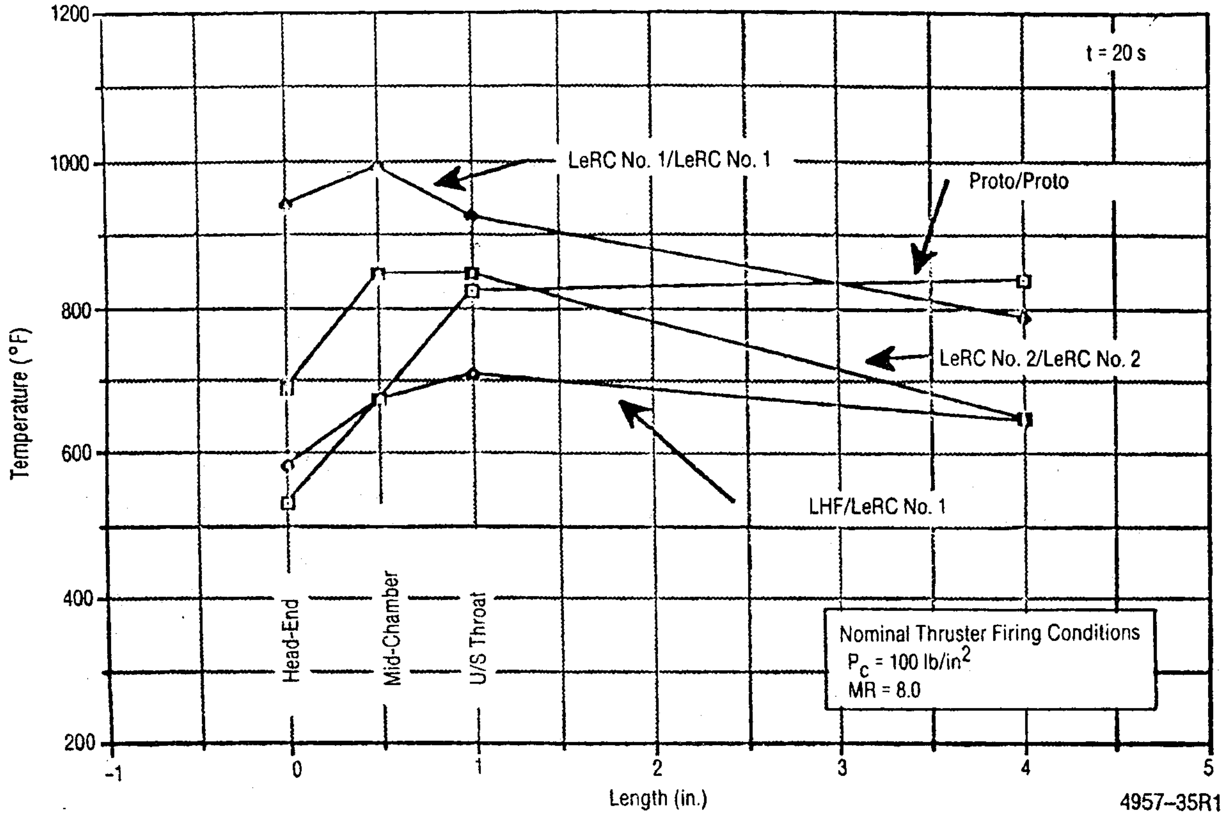


Figure 4-20. 25-lbf GO_2/GH_2 Thruster Comparison of Temperatures

Circumferential variations in measured surface temperatures were also observed. Figure 4-21 displays measured results at the mid-chamber location for the prototype injector/LeRC 2 thrust chamber, the LeRC 1 and 2 thruster and LHF injector/LeRC 1 thrust chamber. The thermocouple T6034 locations seemed to be the typical "hot spot" for all configurations.

In an attempt to determine if coolant flow in the thruster or uneven injector distribution was the main cause of the variations, the LeRC 2 unit was assembled and hot fired with the injector rotated 135 deg, with respect to the thrust chamber. Figure 4-22 displays the measured temperature distribution obtained and, for comparison, the results from the normally assembled unit. The circumferential variations were reduced, but the T6034 locations continued to read the highest temperature. The conclusion was that both the

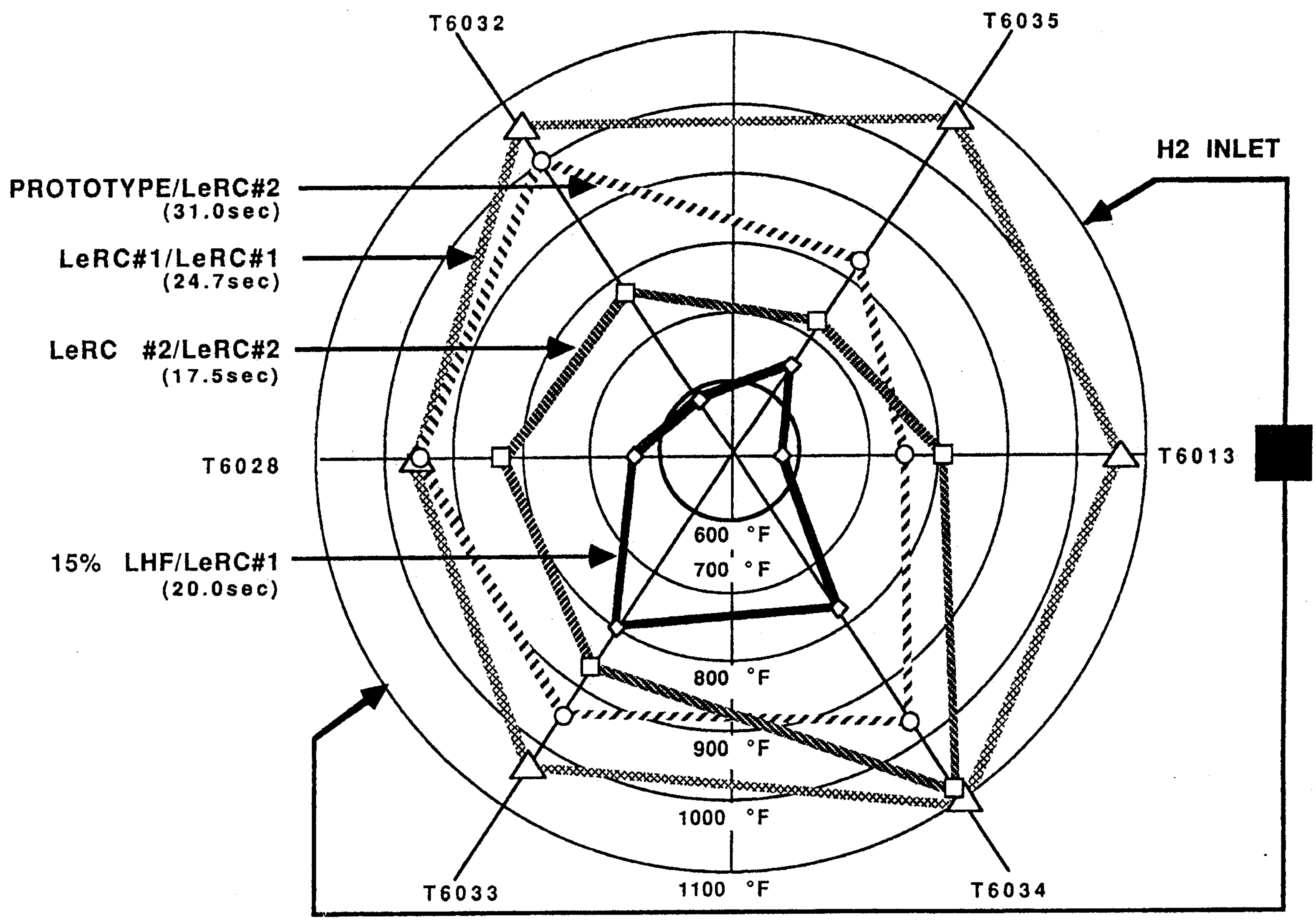


Figure 4-21. Chamber Temperature Distributions

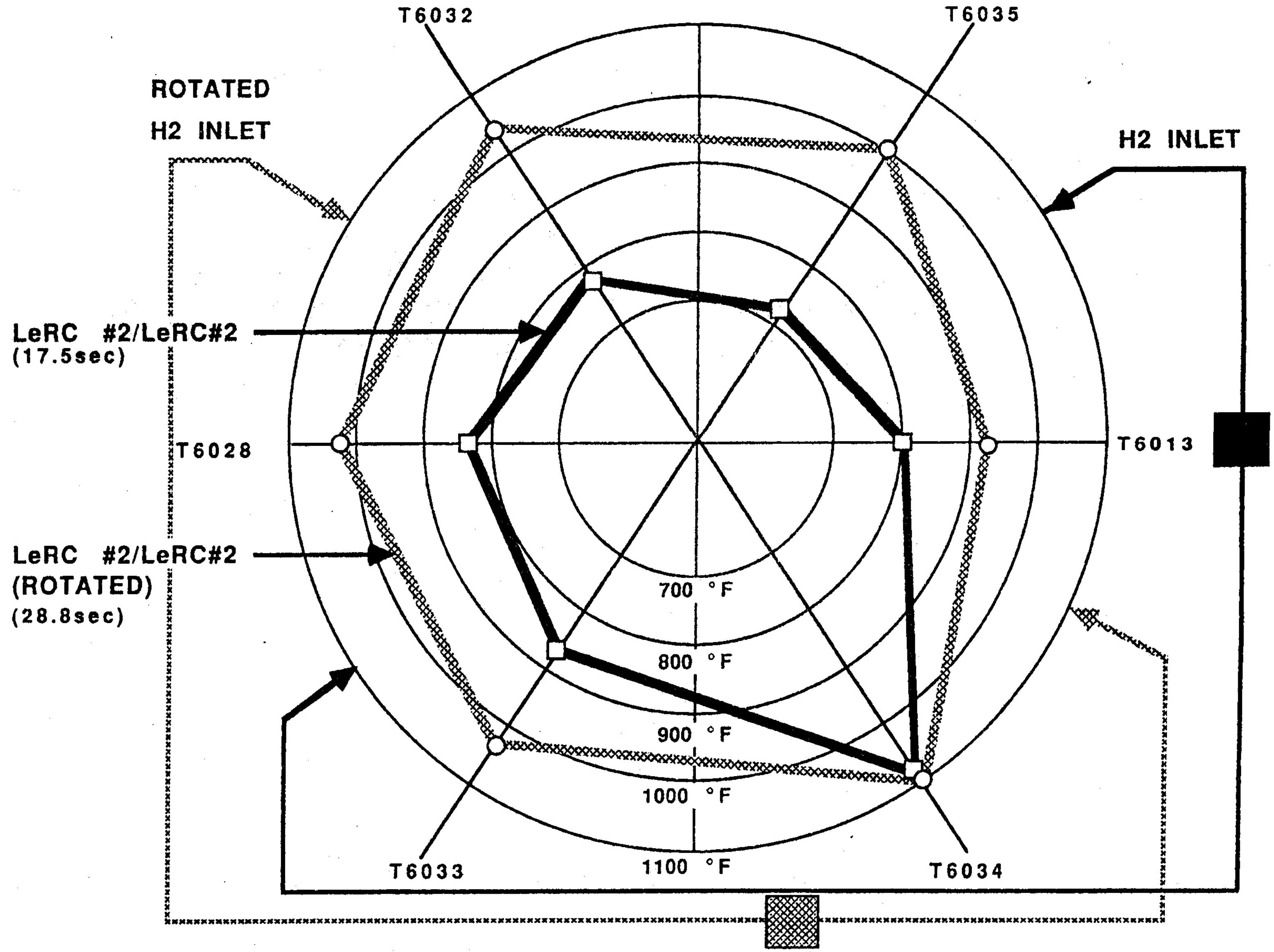


Figure 4-22. Chamber Temperature Distributions - LeRC 1

injector and thrust chamber had uneven flow characteristics. The extent of the maldistribution could not be quantified from these data.

Evidence of flow maldistribution of the injector was seen in the flame pattern on the interior surfaces of the thrust chamber. Figure 4-23 is a photo of the inner thrust chamber walls looking toward the throat, and displays the flame pattern observed. The marks are discolorations only and would disappear if the chamber was used with a different injector. No detectable erosion was observed on any of the thrust chambers.

To assess flow distribution, a cold-flow program was instituted to measure the flow from each injector element. The following section discusses such tests and the results obtained.

4.1.3.5 <u>Cold-Flow Testing</u>. To measure the flow emanating from each injector element, two types of laboratory experiments were performed in the Rocketdyne Engineering Development Laboratory: water flow and gaseous flow.

The water flow tests were performed with the injector flowing at ambient pressure and at very low flow rates (pressure drop) to preclude cavitation. Flexible tubing was held over each orifice in turn, and the flow collected for a specified time (usually 1 min). Both fuel and oxidizer circuits were tested in this manner.

The gaseous flow tests were performed with gaseous nitrogen. The flow from each element was collected with small, flexible tubing held over each orifice in turn, as in the water flow experiments. The flow was measured by a calibrated flow ball manometer. Pressure drops of 5 to 10 psig were maintained during the tests.

Figure 4-24 displays the results of the water and gaseous flow tests for each oxidizer injector element as a percentage of total injector oxidizer (simulated) flow. Reasonable agreement is apparent between the water and gaseous flow tests. The flame pattern, traced from Figure 4-23, is also

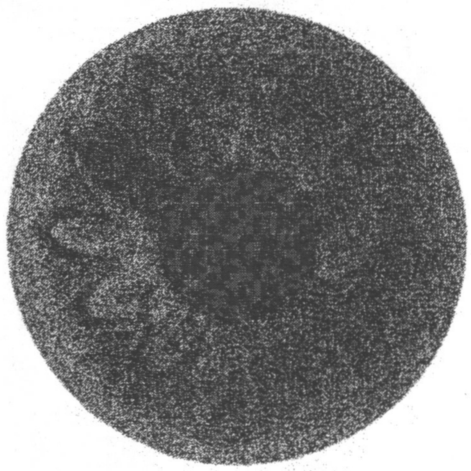


Figure 4-23. LeRC 25-lbf GO_2/GH_2
Thruster Flame Patterns
(1XA25-10/28/87-C1E*)

shown. Correlation to the flow distribution patterns was considered sufficient to attribute the flame "streaking" pattern to the oxidizer flow distribution. The "streaks" for elements 5 through 10 correspond to high flow results from the water flow testing.

Attempts were made to calibrate or redistribute the flow by reaming or scraping the interior of the oxidizer tube in the area of the two 0.043-in. holes in the upper end of the oxidizer post (see Appendix A, drawing 7R032629). As shown in Figure 4-25, the results were not successful. Recalibration of the LHF injector to improve the flow distribution was quite successful, however. Figure 4-26 displays the results of the efforts to recalibrate and indicates the flow from each element was adjusted to within ±3% of the overall average.

Calibration of individual oxidizer posts in a fixture prior to assembly into an injector is recommended for any future units to preclude oxidizer flow maldistribution into the combustor.

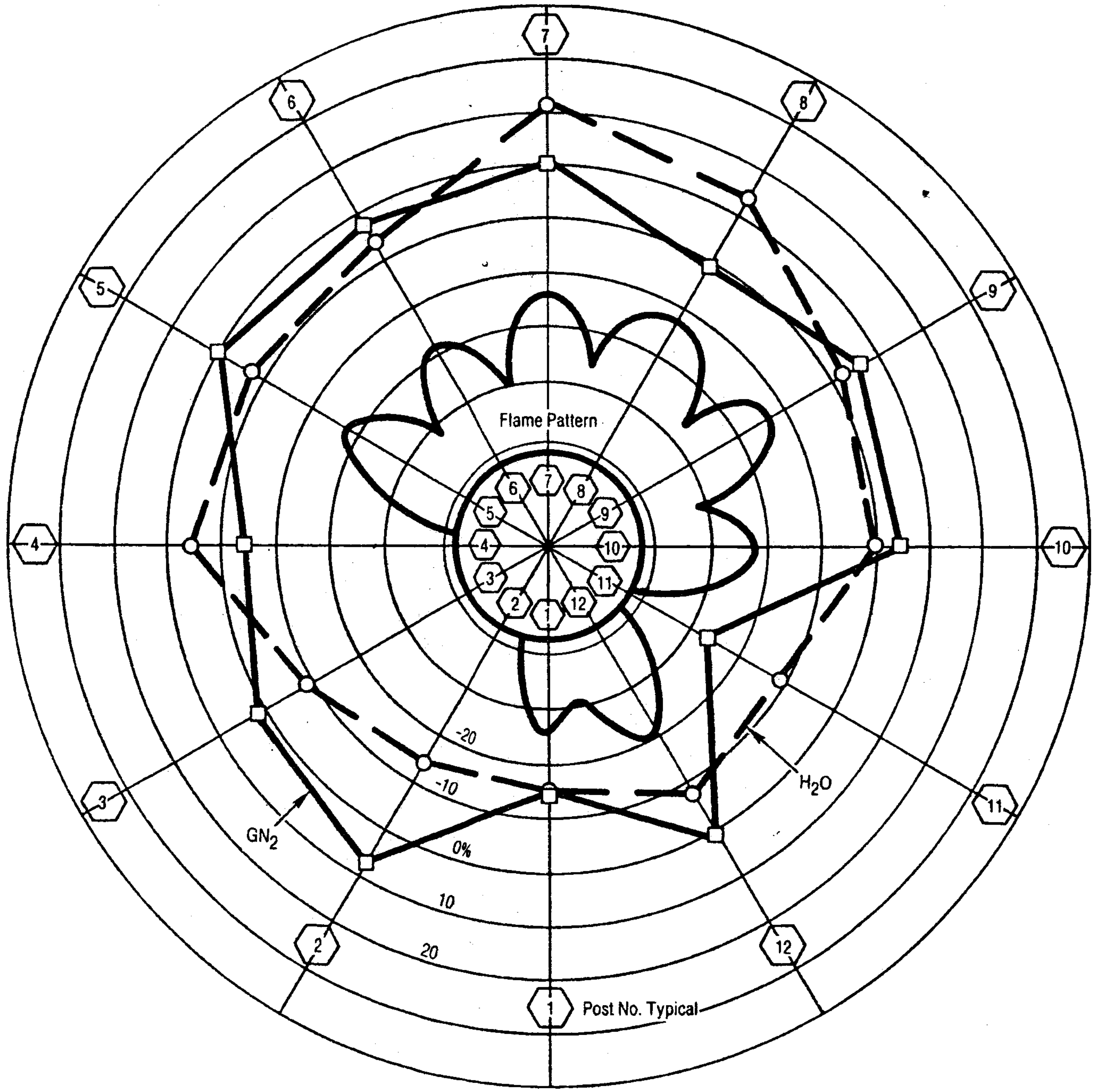


Figure 4-24. Element-by-Element Cold Flow Distribution Comparison to Flame Pattern

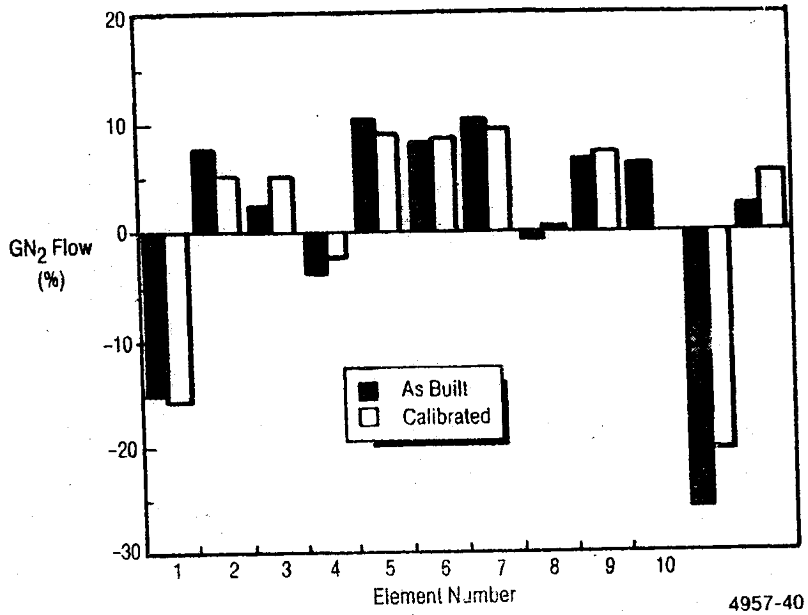


Figure 4-25. Oxidizer Post GN_2 Cold-Flow Test LeRC 1 Injector

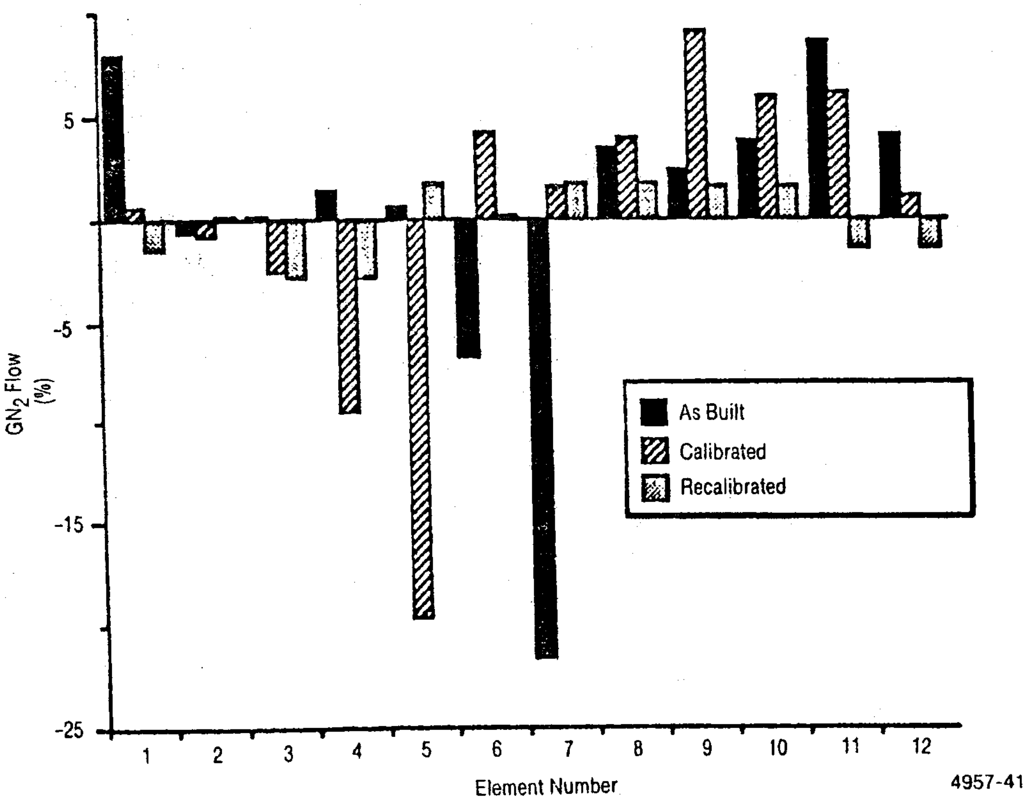


Figure 4-26. Oxidizer Post GN_2 Cold-Flow Test, Low-Heat-Flux Injector

4.1.4 Thruster Operating Regime

The parameters governing the allowable operating conditions of the thruster are the thrust chamber wall temperature (combustion gas side) and the wall temperature differential from the combustion side to the back wall (coolant side). The wall temperature differential will not vary significantly as chamber steady-state operating conditions are changed (chamber pressure and mixture ratio). The high conductivity of the NARloy material used for the thrust chamber liner tends to level or smooth any temperature variations.

The external thrust chamber temperature measurements provide a reasonable indication of the inside wall temperature and are used to establish "redline" parameters for testing. These data and the thrust chamber heat transfer characteristics, described above and in section 3.1.1, were used to establish safe and marginal operating regimes for the prototype and LeRC thrusters and for the thruster using the low-heat-flux injector. The results of a study are shown in Figure 4-27, which portrays the safe operating regime as a function of chamber pressure and mixture ratio. Data points for full duration (30 s+) runs and for runs terminated by the external thrust chamber temperature exceeding the established redline values are shown for reference.

Typical thruster operating parameters are presented in Table 4-5 for oxygen and hydrogen temperatures and pressures at the thruster inlet and at key locations throughout the thruster assembly. These are typical temperature and pressure readings and are considered to be representative of the two delivered units. Values of the parameters predicted for use in future applications are also shown for comparison. Particular causes for variations from measured values are discussed in Section 3.1. It is felt that, with corrected fabrication procedures, the predicted pressure values are satisfactory for use in future applications.

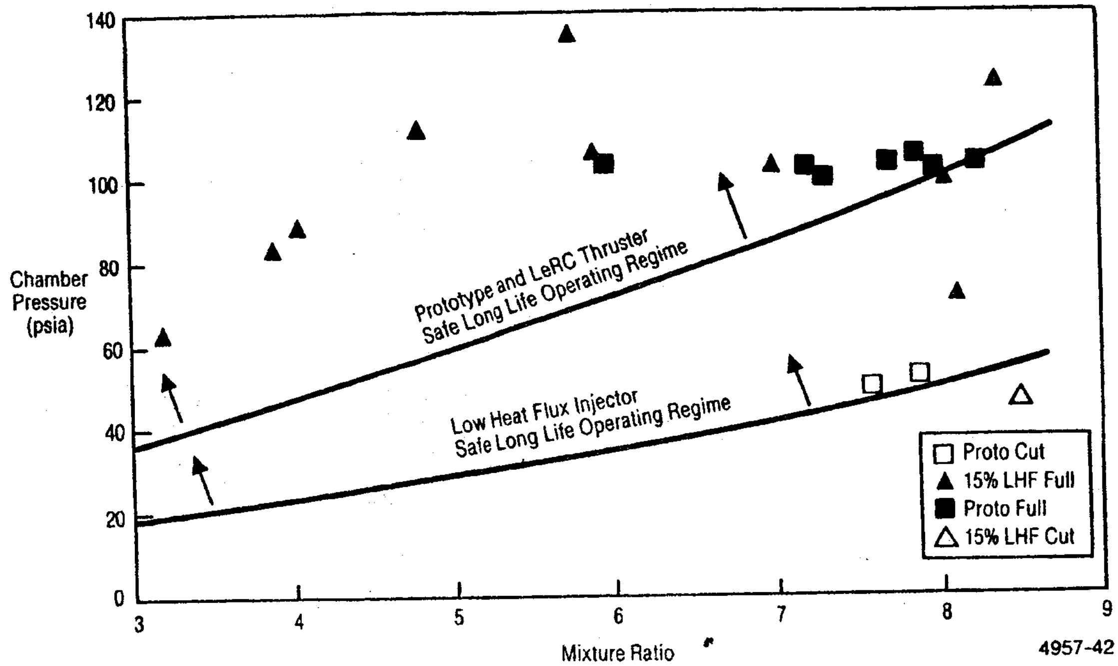


Figure 4-27. 25-lbf GO_2/GH_2 Thruster Safe Long-Life Operating Regime

Table 4-5. Typical Thruster Operating Parameters

Location	Oxidizer				Fuel			
	Predicted		Typical Measured		Predicted		Typical Measured	
	Pressure	Temperature	Pressure	Temperature	Pressure	Temperature	Pressure	Temperature
Thruster inlet	166	70	167	70	177	70	259	70
Thrust chamber inlet	NA	NA	NA	NA	173	NA	255	70
Injector inlet	139	150	140	170	127	950	120	850
Chamber	100	NA	100	NA	100	NA	100	NA

5.0 CONCLUSION

The 25-lbf thrust oxygen/hydrogen thruster operation for the Freedom Station was demonstrated, and two thruster assemblies were delivered to NASA-LeRC for further demonstration testing and evaluation.

The thruster design was based on the successful prototype unit developed during the Freedom Station Phase B studies as the result of Rocketdyne company-funded effort. Although the prototype thruster was originally designed for operation at a propellant mixture ratio of 4, interim modifications were made to the chamber and injector to provide increased cooling capability and thereby extend the operational range to a mixture ratio of 8 (stoichiometric). The temporary modifications proved to be successful in demonstrating long-term operational capability at the extended conditions.

For the current program, the high mixture ratio design modifications were incorporated to adapt the basic design to operate throughout a mixture ratio range of 3 to 8 in accordance with program requirements. A summary of the modifications to the thruster assembly is presented in Table 3-1.

Over 100 tests were conducted during the current program to provide data for performance optimization and characterization of the modified thruster assemblies. When combined with previous exhaustive testing of the prototype thruster, this 25-lbf thrust oxygen/hydrogen thruster has been tested more extensively prior to design, development, test, and evaluation (DDT&T) than any previous engine or thruster.

Manufacturing producibility of the thruster assembly has been validated extensively during the fabrication of the two delivered units and the previous prototype and experimental hardware. The sensitivity of performance to dimensional tolerances in the hardware components requires careful specification, inspection, and control during the fabrication phase. This is particularly true of chamber coolant channel machining, and injector fuel annulus gap variations and concentricity of the oxidizer injection posts. Recommended tolerances for the latter are presented in Table 3-3.

The deliverable thruster assembly designs were found to meet all program requirements with no limitations or qualifications. To extend this capability even further in recognition of the rigorous, long-life requirements of the Freedom Station flight configurations, additional design evolution should be considered. For example, the incorporation of the low-heat-flux injector will enhance propulsion system reliability and life by a significant reduction in temperature of thrust chamber hardware and hydrogen injection temperatures.

The anticipated safe, long-life operating regime of the thruster with the prototype injector and low-heat-flux injector has been established as a function of chamber pressure and mixture ratio from data generated and is presented in Figure 4-27.

Typical measured values of temperatures and pressures throughout the two delivered thruster assemblies are compared with predicted values of the parameters and presented in Table 4-5. Although there are some sizable variations in pressures, causes have been identified, and the predicted values are satisfactory for use in application studies.

0398e/tab

6.0 REFERENCES

1. Space Station Propulsion Test Bed, Contract NAS3-36418, Rocketdyne, May 1985

2. CPIA Publication 246, April 1975

7.0 BIBLIOGRAPHY

1. L. E. Finden, G. L. Briley, and R. S. Iacabucci, 25-LBF GO2/GH2 Space Station Thruster, Rockwell International/Rocketdyne Division, Canoga Park, CA, July 1988

2. J. F. Glass, W. Tu, S. J. Ebert, and M. J. Adams, Space Station System Computer Modeling, Rockwell International/Rocketdyne Division, Canoga Park, CA, June 1986

3. B. S. Heckert and T. I. Yu, 25-LBF Gaseous Oxygen/Gaseous Hydrogen Thruster for Space, Rockwell International/Rocketdyne Division, Canoga Park, CA, August 1986

4. J. G. Campbell and R. C. Stechman, Water Electrolysis System Testing, AFRPL-TR-74-72, The Marquardt Company, November 1974

5. NASA CR-120805, Hydrogen-Oxygen APS Engines, Volume I: High Pressure Thruster, Contract NAS3-14352, Rockwell International/Rocketdyne Division, Canoga Park, CA, February 1974

6. NASA CR-120806, Hydrogen-Oxygen APS Engines, Volume II: Low Pressure Thruster, Contract NAS3-14352, Rockwell International/Rocketdyne Division, February 1973

7. NASA CR-120869, Hydrogen-Oxygen Catalytic Ignition and Thruster Investigation, Volume I: Catalytic Ignition and Low-Pressure Thruster Evaluation, Contract NAS3-14347, TRW, November 1972

8. NASA CR-230870, Hydrogen-Oxygen Catalytic Ignition and Thruster Investigation, Volume II: Catalytic Ignition and High-Pressure Thruster Evaluation, Contract NAS3-14347, TRW, November 1972

9. NASA CR-120895, Hydrogen-Oxygen Auxilliary Propulsion for the Space Shuttle, Volume I: High-Pressure Thruster, Contract NAS3-14354, Acrojet Liquid Rocket Company, January 1972

10. NASA CR-72972, Space Shuttle Auxiliary Propulsion System (APS) Ignition System, Contract NAS3-14352, Rockwell International/Rocketdyne Division, May 1971

11. NASA CR-120976, Space Shuttle Auxiliary Propulsion System Valves, Contract NAS3-14350, Rockwell International/Rocketdyne Division, June 1973

12. Schmidt, G. R., "The Impact of Integrated Water Management on the Space Station Propulsion System," Contract NAS8-36526, Booze-Allen and Hamilton, Inc., Washington, D.C., July 1987

0398e/bjb

APPENDIX A
DRAWINGS

Included in this section are the drawings of the 25-lbf thruster and all component parts. A complete list of the drawings is displayed in Table A-1.

0398e/bjb

TABLE A-1. LeRC 25-lbf GO_2/GH_2 Thruster Assembly

Part Number	Description	Figure Number
7R033601	Thruster - assembly of 25 lb GO_2/GH_2	A-1
7R033603-1	Nozzle assembly 25-lb GO_2/GH_2	A-2
7R033603-3	Nozzle machine contour	
7R033603-5	Manifold ring	
7R033603-7	Flange	
7R033603-9	Closure	
7R033603-13	Nozzle	
7R033657	Injector - Coaxial downpass, assy of	A-3
7R033657-3	Tubing	
7R033657-9	Body	
7R032648-1	Fitting injector downpass	A-4
7R032648-3	Tee	
7R032648-5	Fitting	
7R032648-7	Ring fuel	
7R032648-9	Ring ox	
7R032648-13	Tube	
7R03262907	Oxidizer post	A-5
7R03263005	Injector face plate	A-6
7R033607-17	Standoff	
7R033607-19	Standoff	
7R033607	Tubing oxidizer and fuel	A-7
7R033607-3	Fuel return	
7R033607-5	Fuel return	
7R033607-7	Fuel feed	
7R033607-9	Fuel feed	
7R033607-13	Oxidizer	
7R033607-15	Fuel	
18001-1	Valve, solenoid 25-lb thrust	A-8
29330	Valve assembly	A-9
7R033602-1	Bracket assembly	A-10
7R033604	Posts, valve standoff	A-11
7433604-3	Post, ox valve	
7433604-5	Post, fuel valve	
7R033605-3	Bracket, valve support	A-12
7R033656-3	Plate	A-13
7R035388-1	Spark plug assy, modified	A-14

0398e/tab

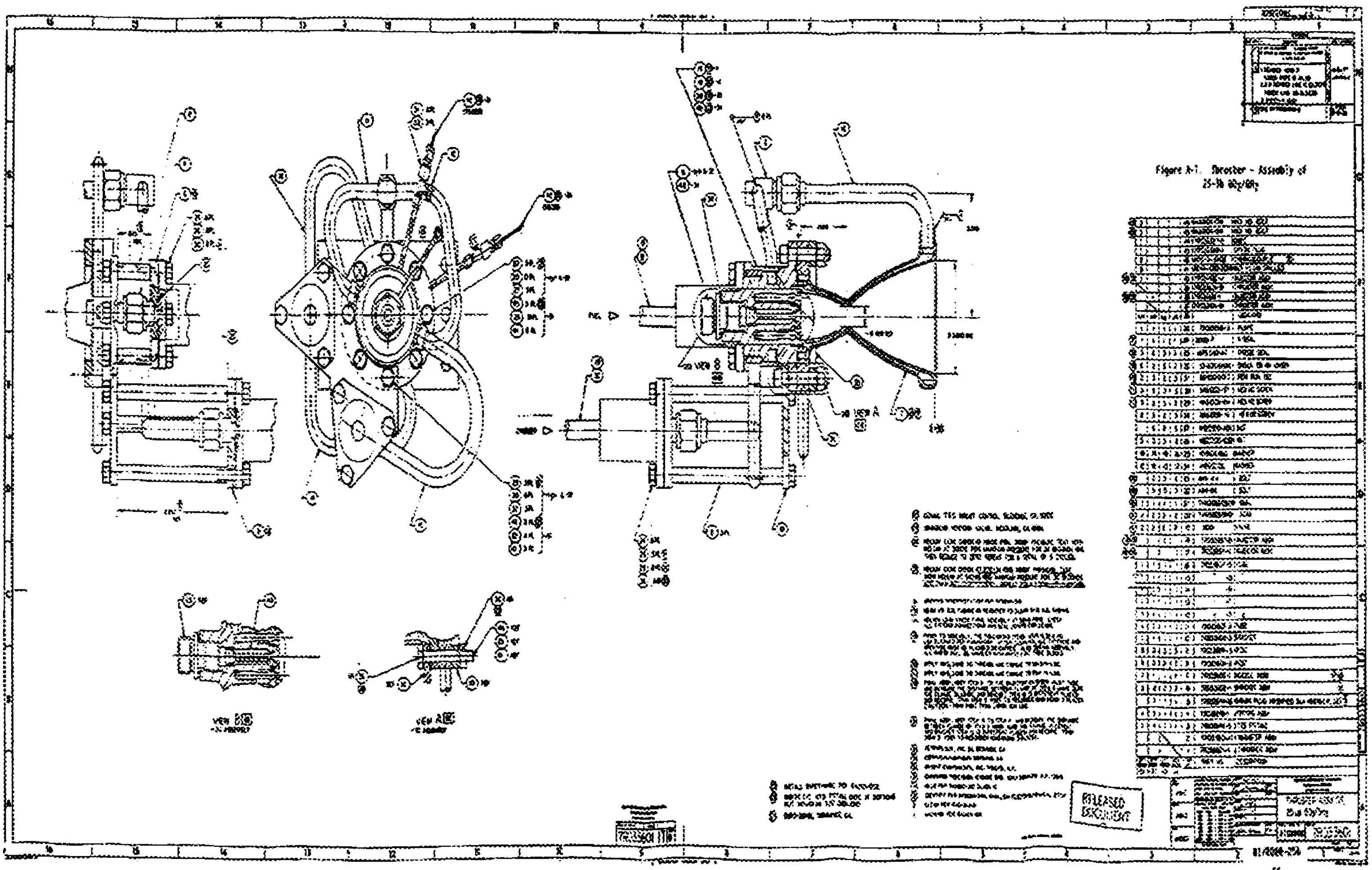


Figure A-1. Thruster - Assembly of 25-lb OF_2/B_2H_6

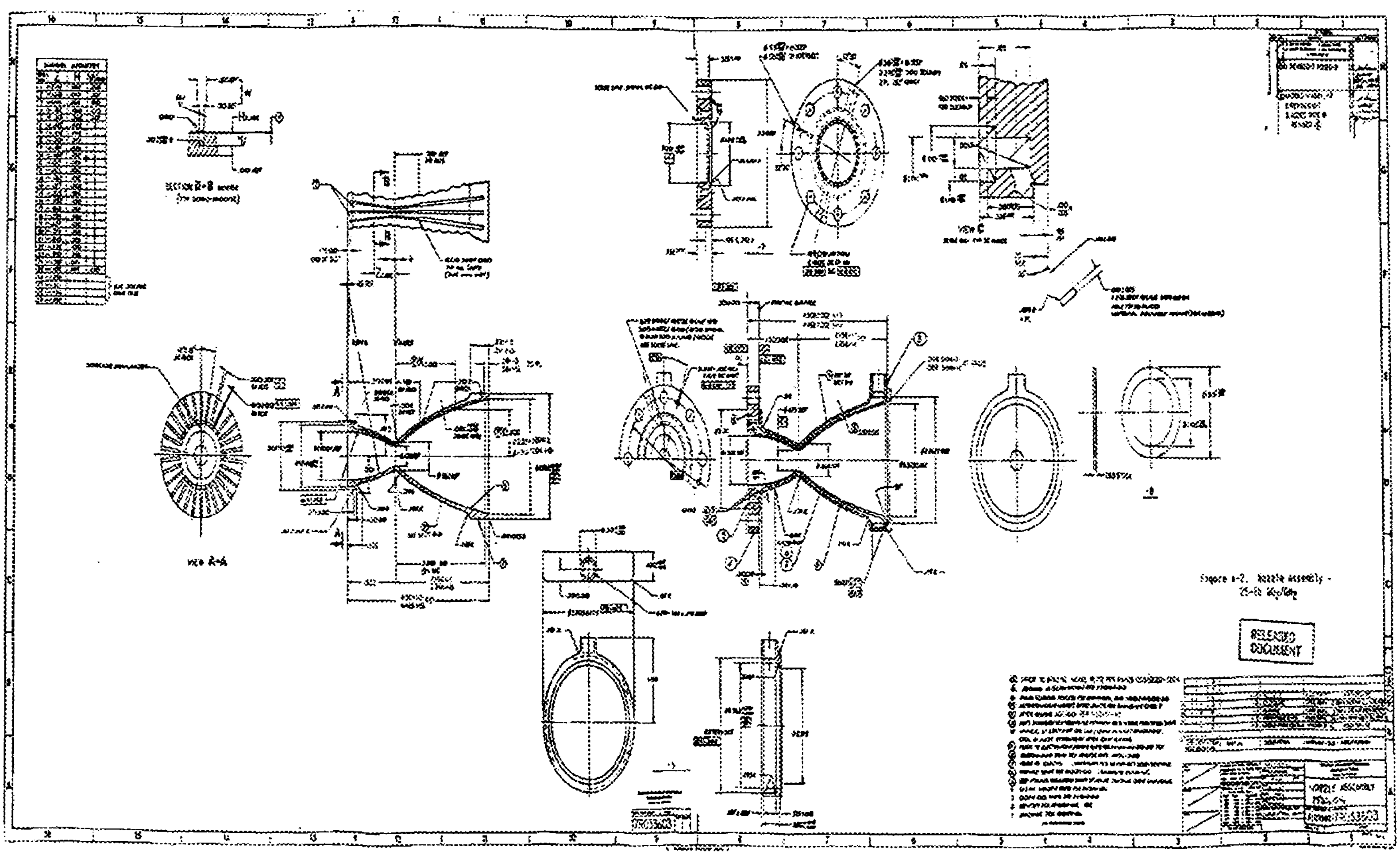


Figure A-2. Nozzle Assembly - 25-lb [illegible]

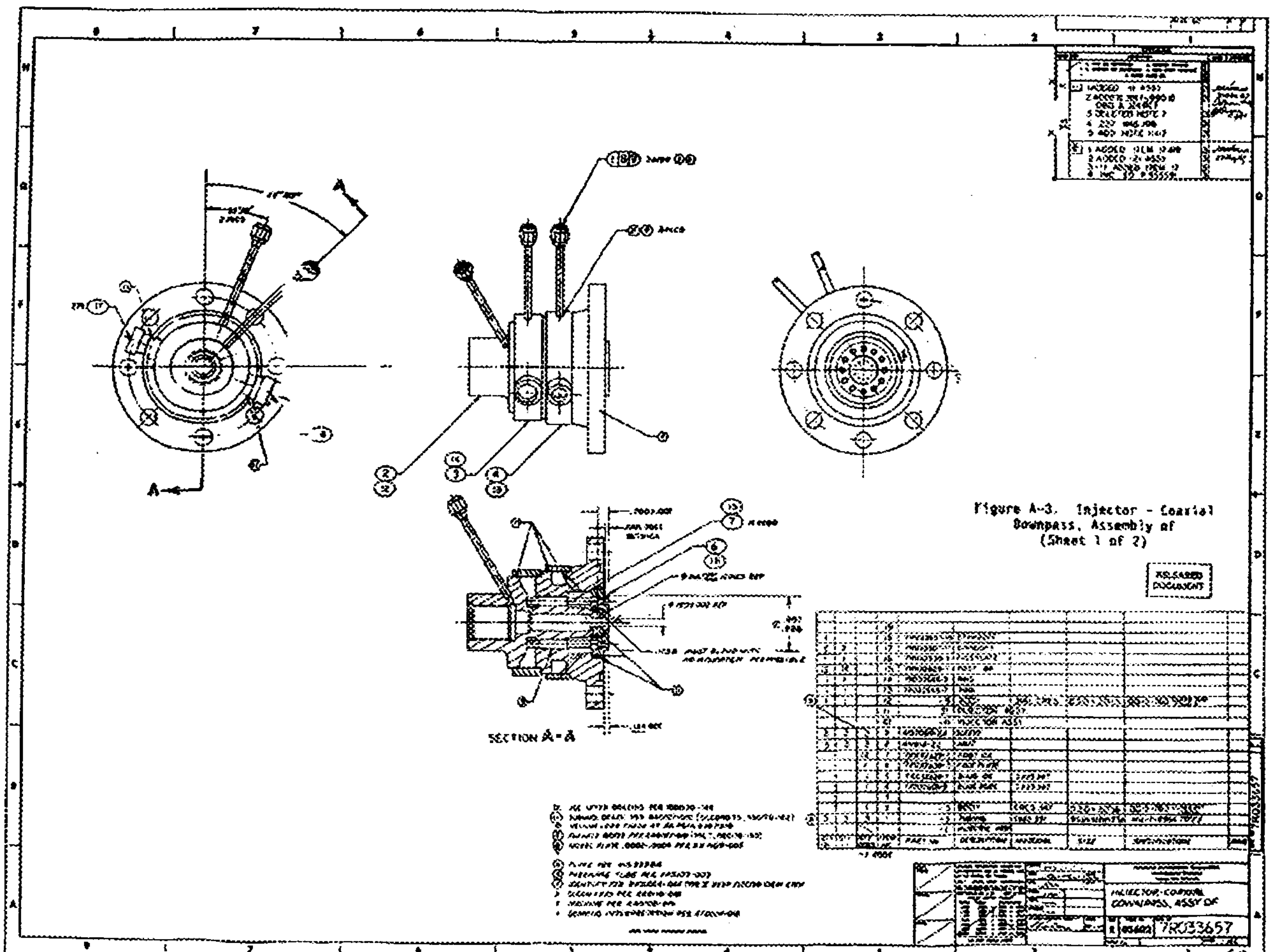


Figure A-3. Injector - Coaxial Downpass, Assembly of (Sheet 1 of 2)

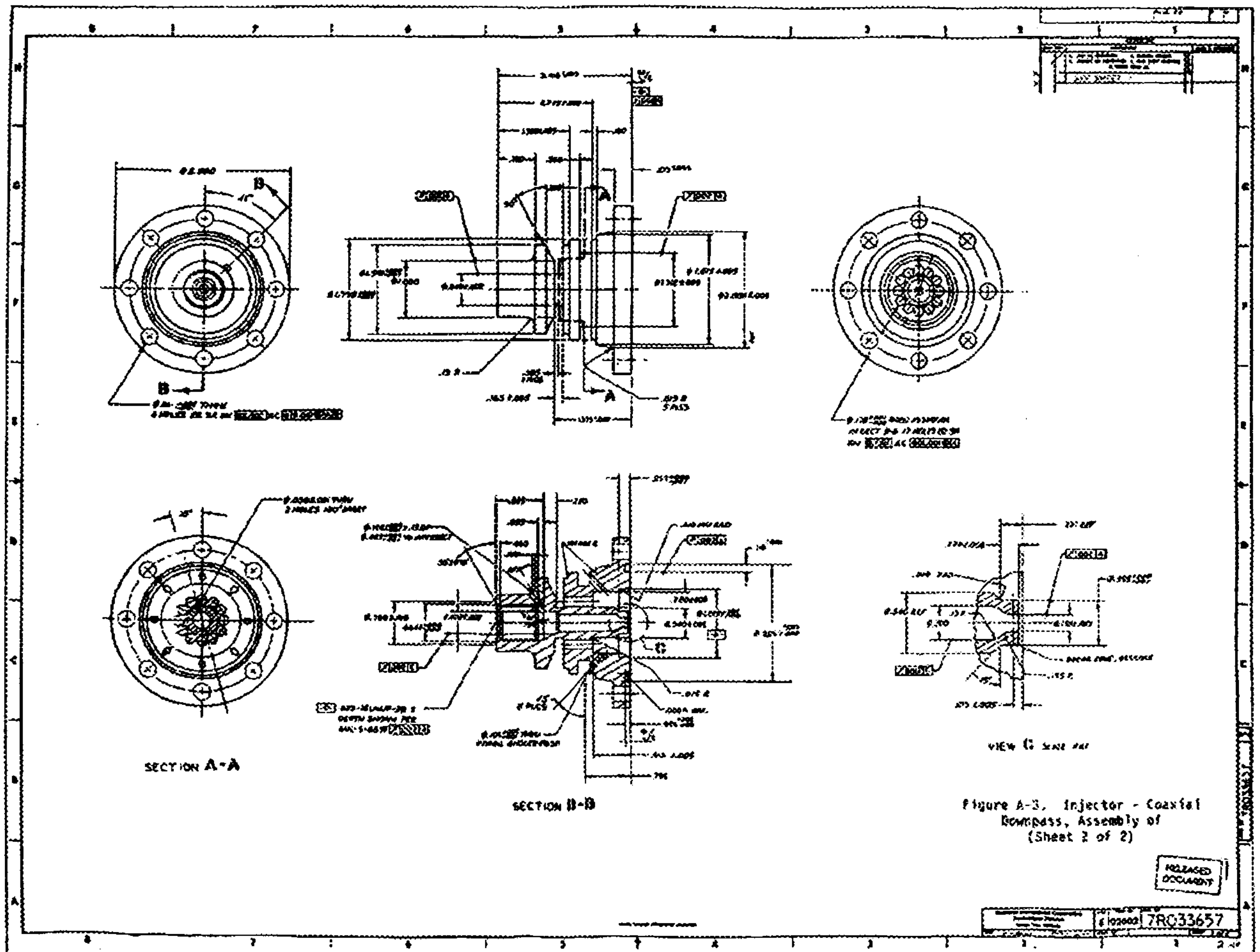


Figure A-3. Injector - Coaxial Downpass, Assembly of (Sheet 2 of 2)

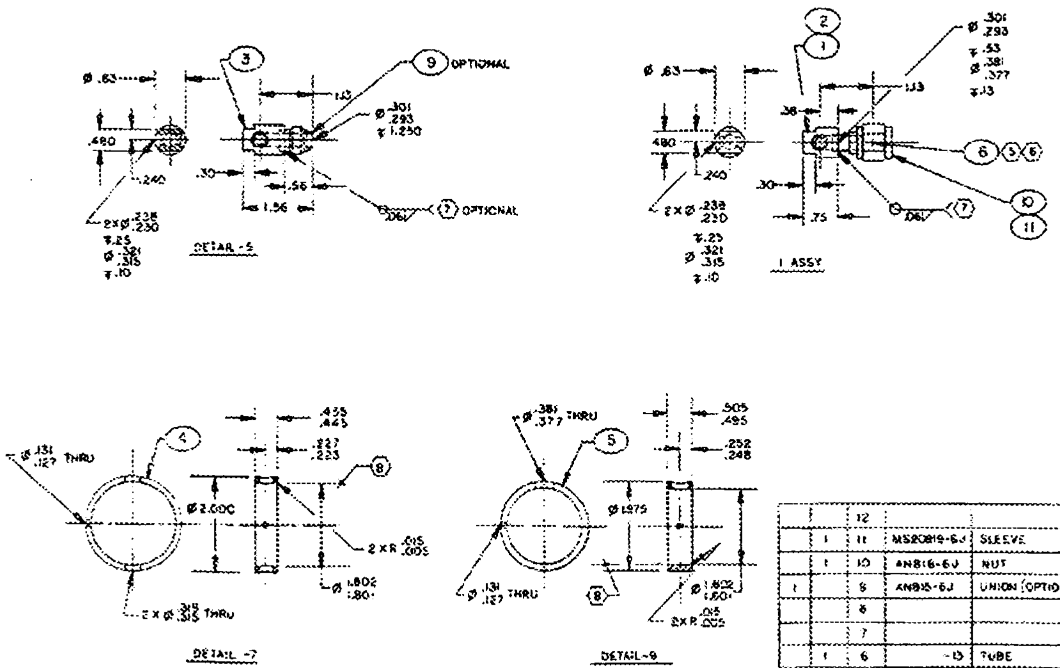


Figure A-4. Fitting Injector Downpass

RI/RD88-256

83

NO REQD	-1 REQD	ITEM	PART NO	DESCRIPTION	MATERIAL	SIZE	SPECIFICATION	ZONE
		12						
	1	11	MS20819-6J	SLEEVE				
	1	10	AN816-6J	NUT				
1		9	AN815-6J	UNION (OPTIONAL)				
		8						
		7						
	1	6	-13	TUBE	321 CRES	3/8OD X.035 X1.0	MIL-T-8808	
1		5	-9	RING OX	316L CRES	Ø2.0 X .6	QQ-S-763 CLASS 316L COND A	
1		4	-7	RING FUEL		Ø2.0 X .5		
1		3	-5	FITTING		Ø.63 X 1.6		
	1	2	-3	TEE	316L CRES	Ø.63 X 1.0	QQ-S-763 CLASS 316L COND A	
		1	-1	FITTING ASSY				

8. NICKEL PLATE PER RA0105-005 THICKNESS .0002-.0004
7. WELD PER RA0607-102 CLASS II
6. FLARE PER MS33584
5. FABRICATE TUBE PER RA0103-005
4. DRAWING INTERPRETATION PER RF0004-018
3. IDENTIFY PER RF0004-044 TAG
2. CLEAN PER RA0110-018
1. MACHINE PER RA0105-016

NOTE: UNLESS OTHERWISE SPECIFIED

Rockwell International Corporation
Rocketdyne Division

FITTING
INJECTOR DOWNPASS

D 02602 7R032648

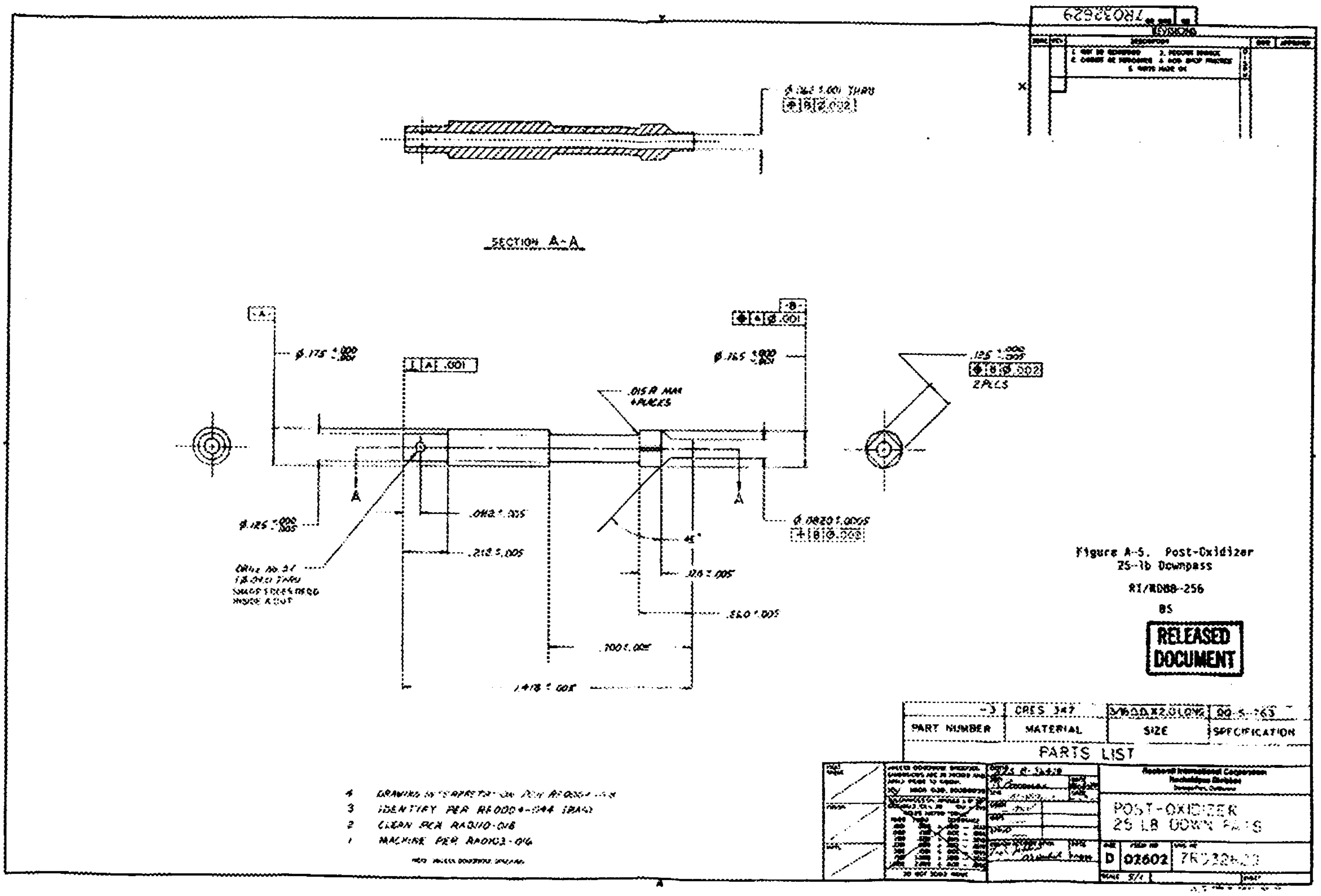


Figure A-5. Post-Oxidizer 25-lb Downpass

RI/RD88-256

85

RELEASED DOCUMENT

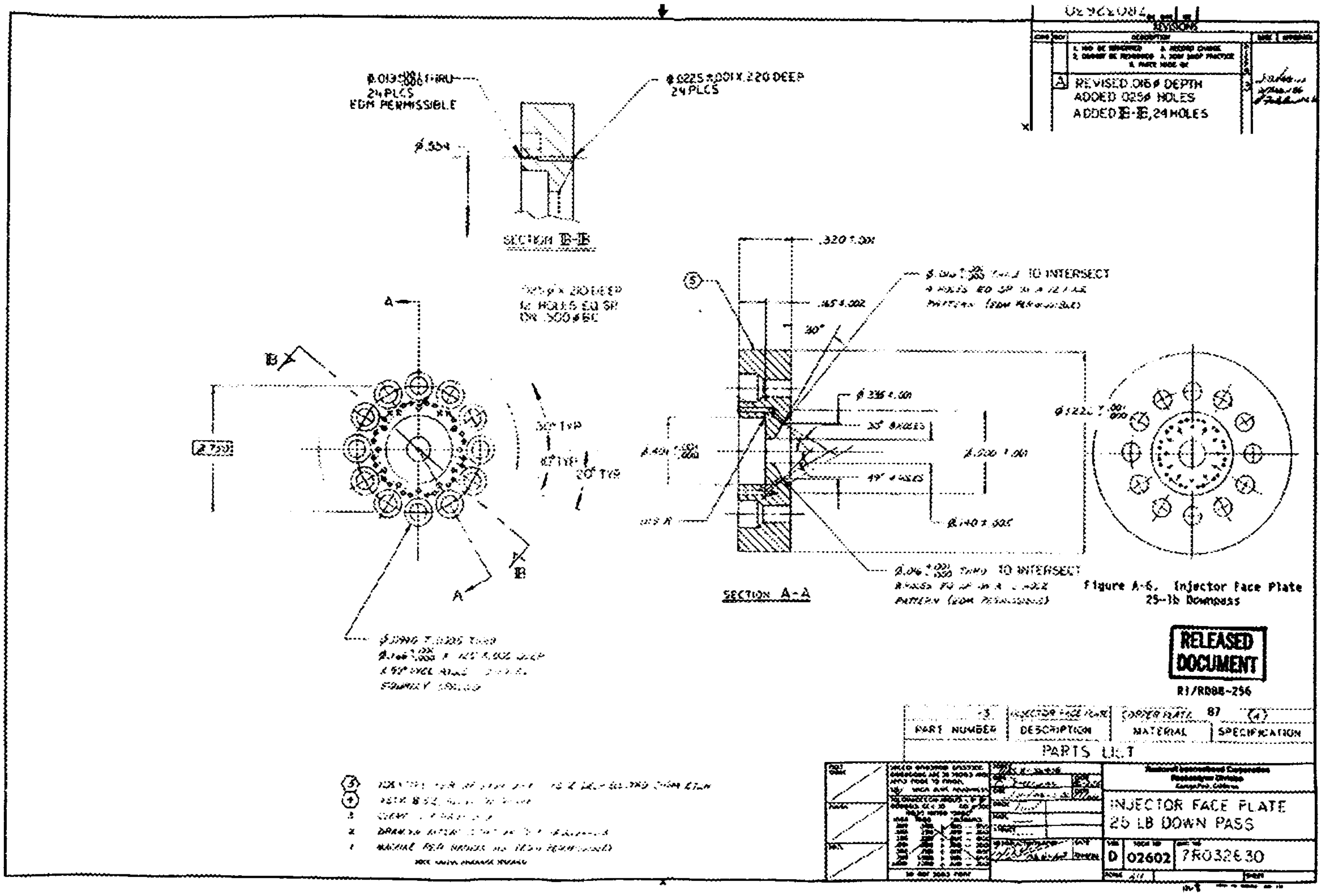


Figure A-6. Injector Face Plate 25-lb Downpass

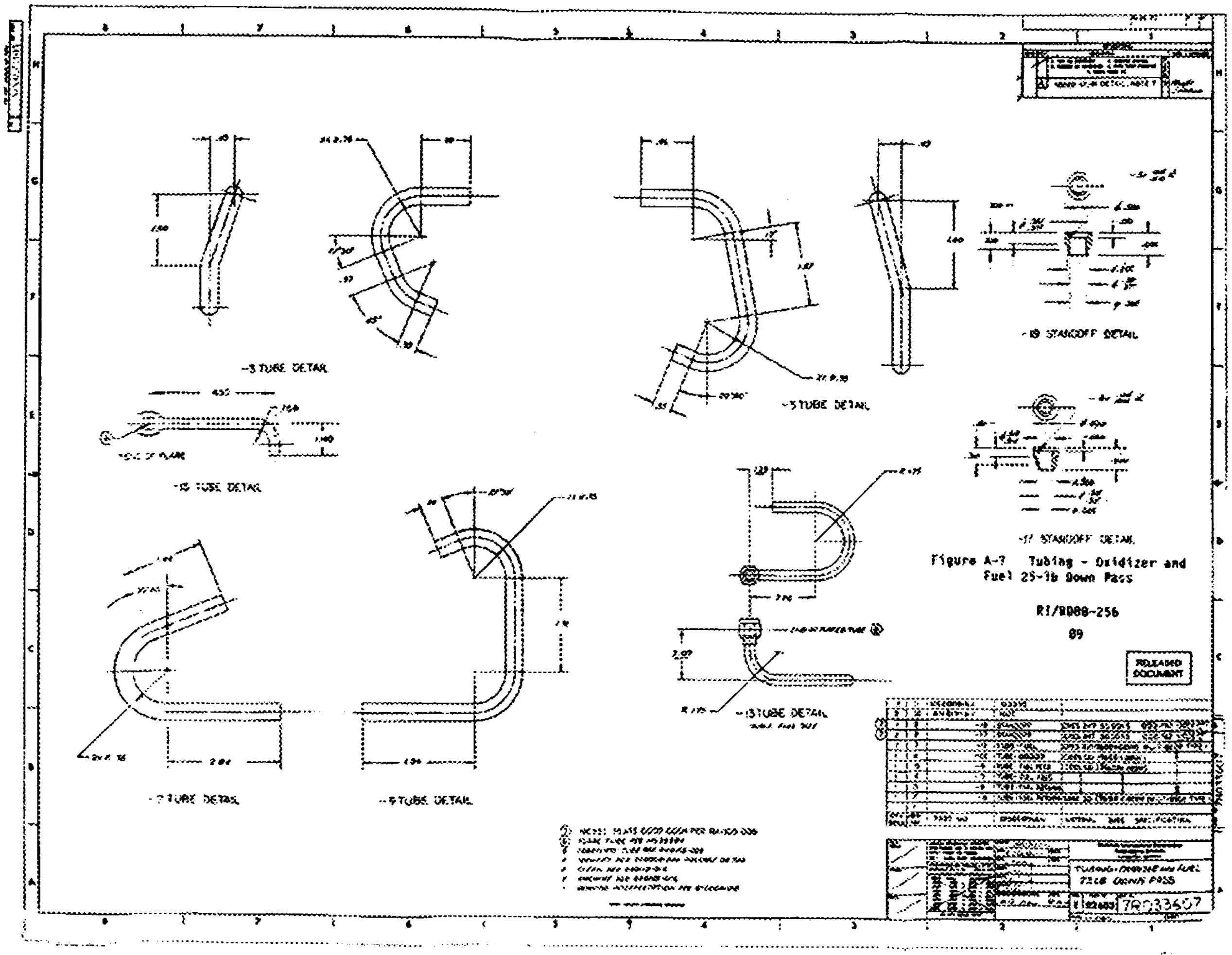


Figure A-7 Tubing - Oxidizer and Fuel 25-lb Down Pass

RI/RD88-256

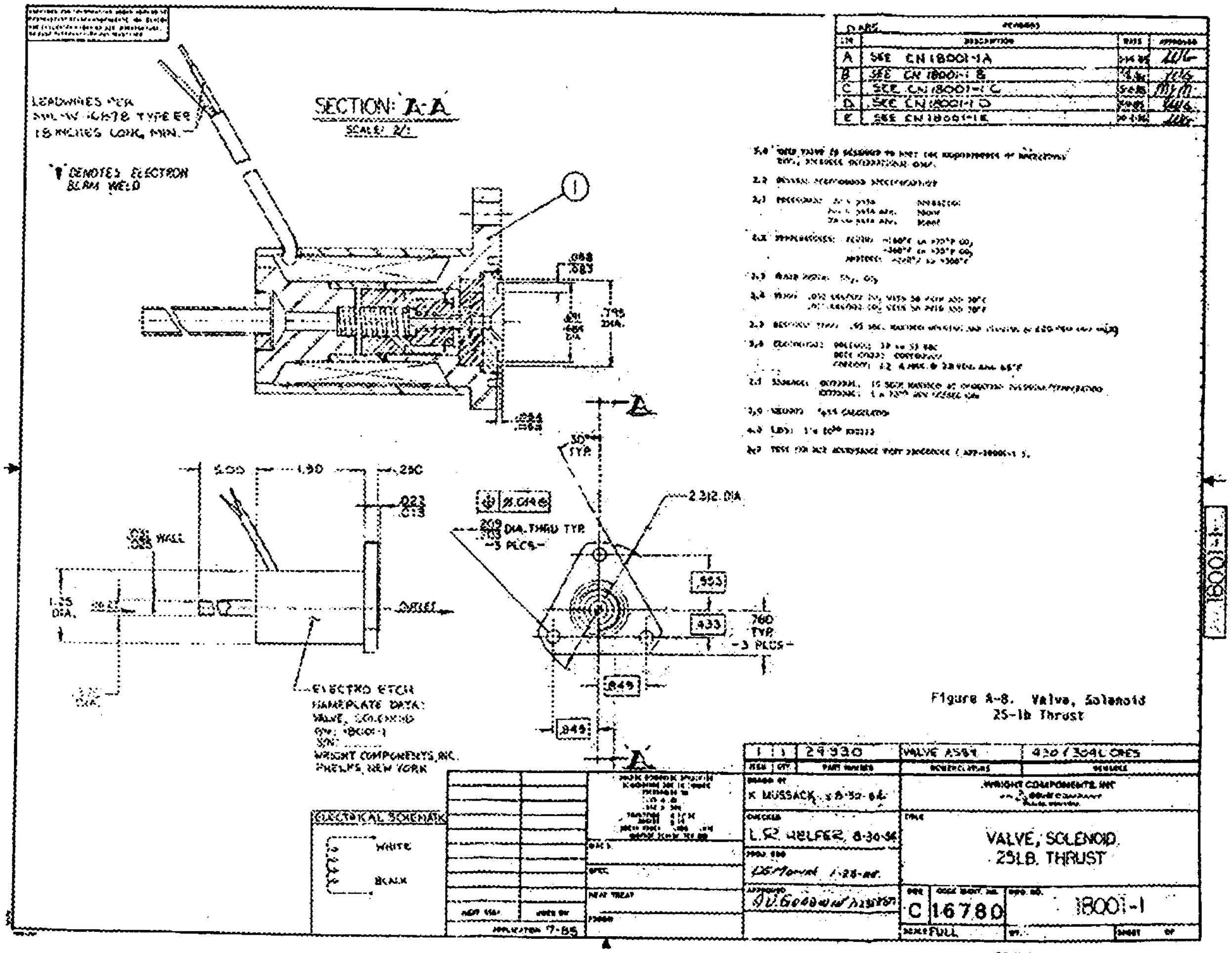


Figure A-8. Valve, Solenoid 25-lb Thrust

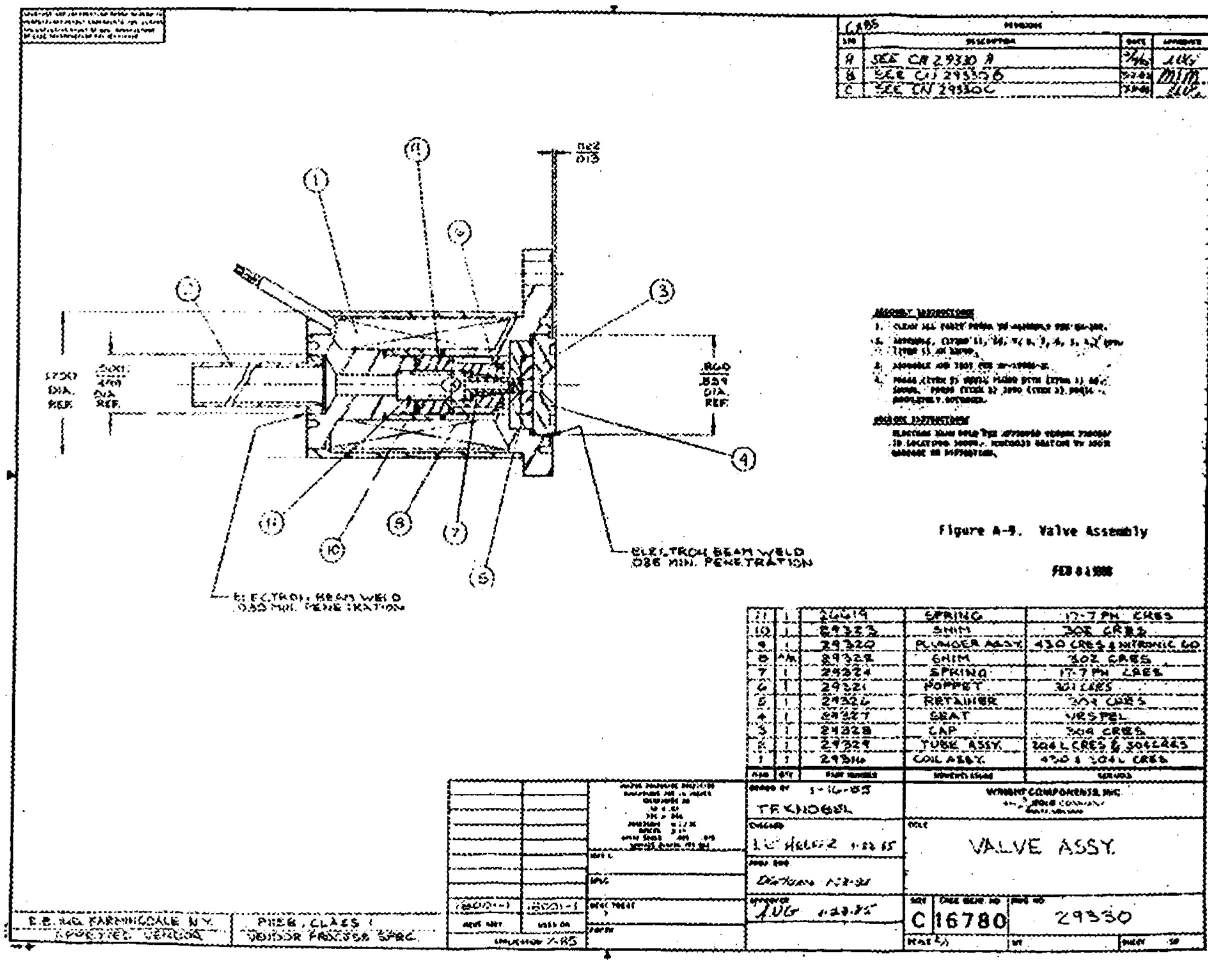


Figure A-9. Valve Assembly

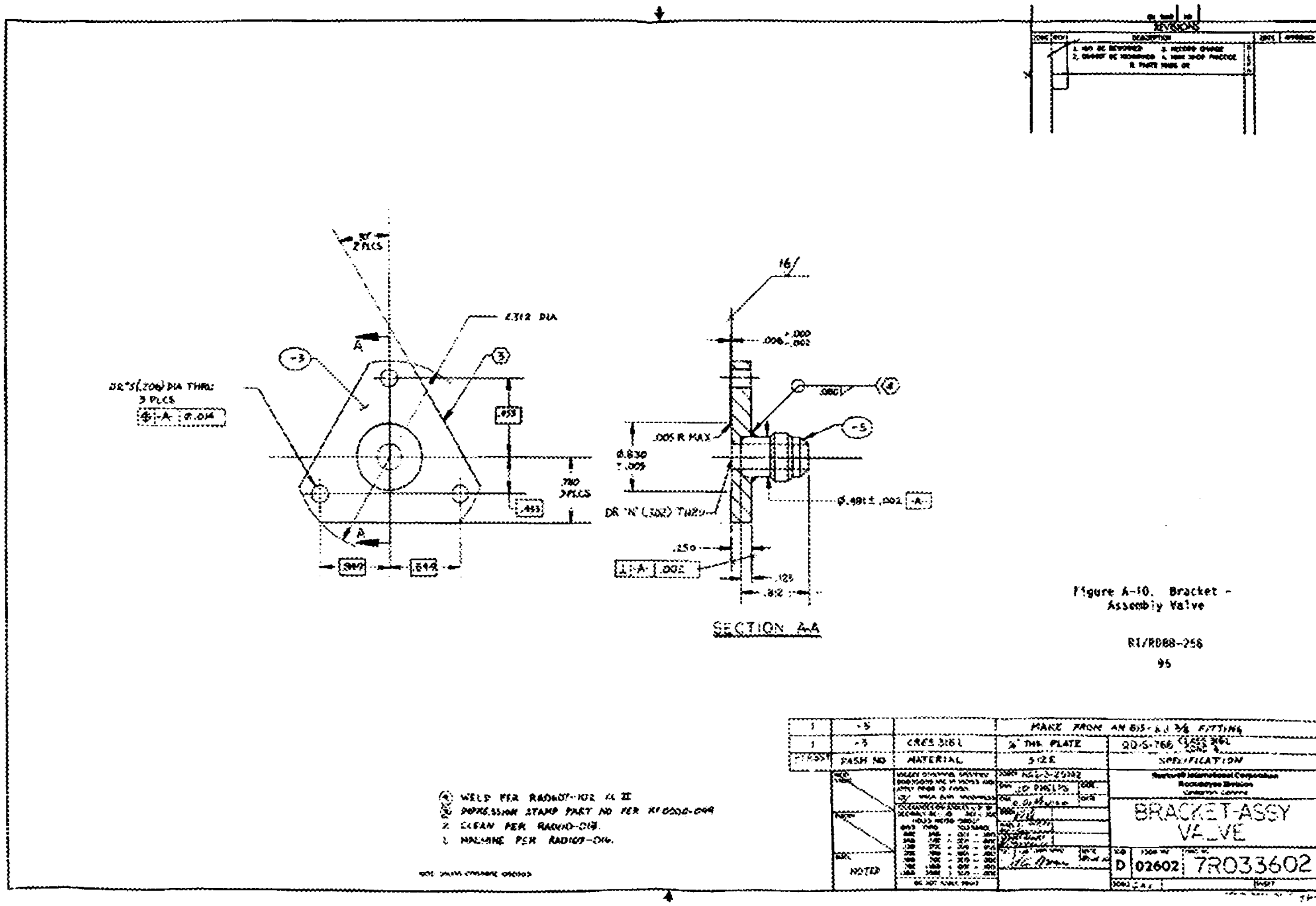


Figure A-10. Bracket - Assembly Valve

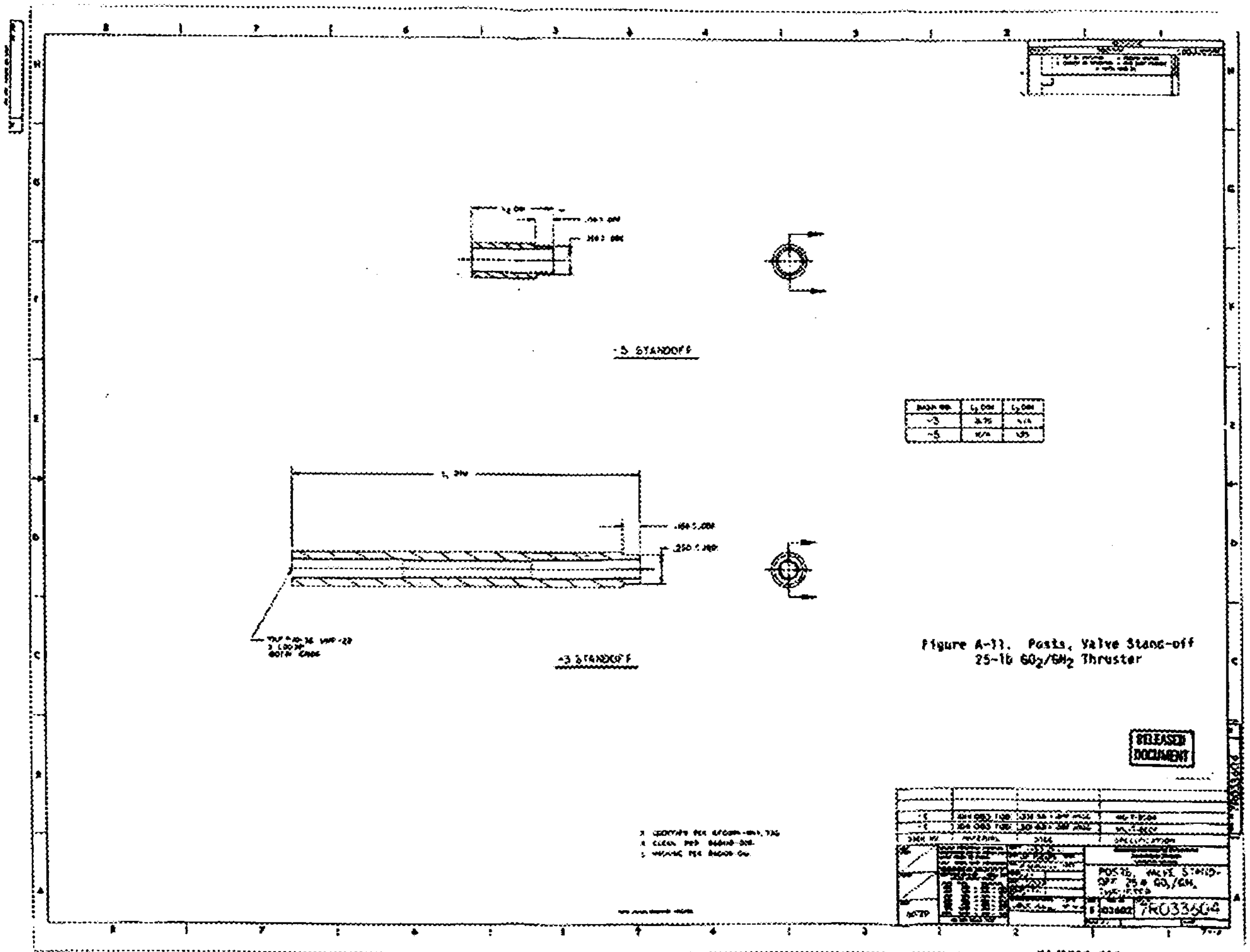


Figure A-11. Posts, Valve Stand-off
25-lb GO_2/GH_2 Thruster

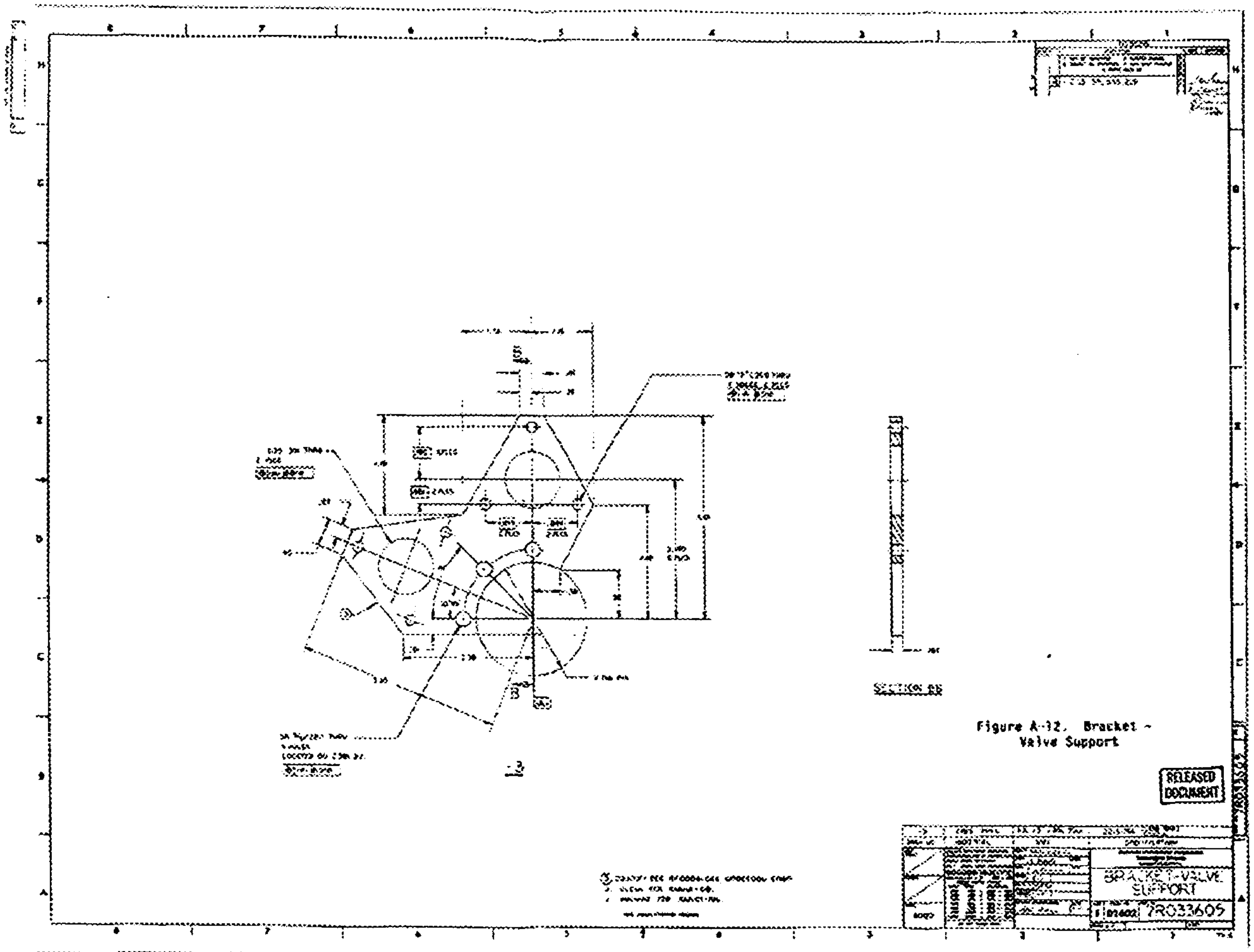


Figure A-12. Bracket - Valve Support

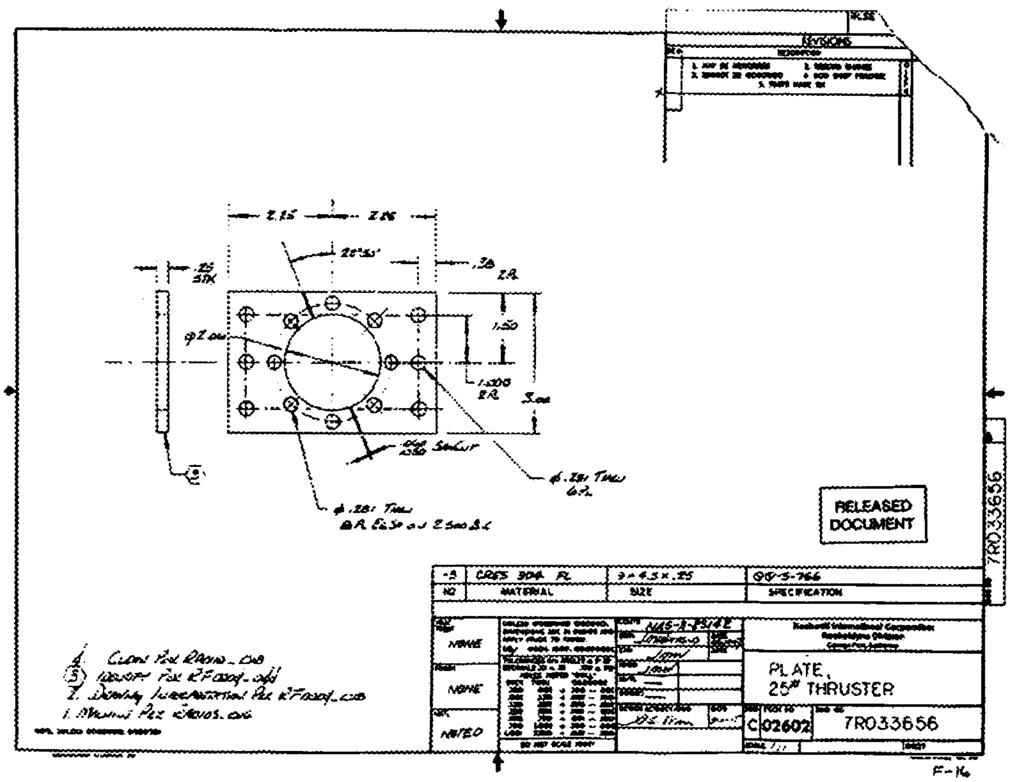


Figure A-13. Plate, 25-lb Thruster
25-lb SO_2/SH_2 Thruster

MAKE FROM 7R035388

REVISIONS

NOTES: UNLESS OTHERWISE SPECIFIED

1. MACHINE PER RA0105-016.
2. CLEAN PER RA0110-018.
3. WELD PER RA0607-102, CLASS II.
4. IDENTIFY PER RF0004-044, TYPE II, BAG & TAG.
5. SIMMONDS PRECISION (ENGINE SYS. DIV.) NORWICH, N.Y. 13815.

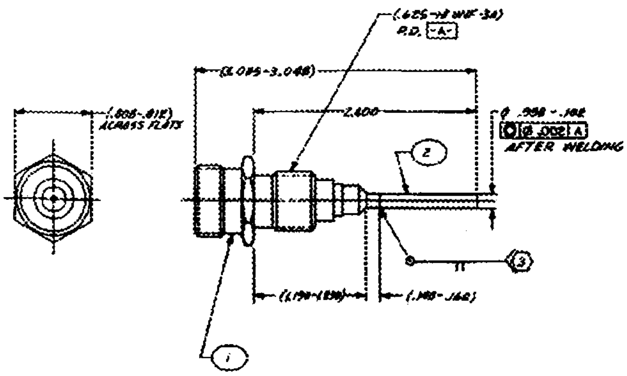


Figure A-14. Spark Plug Assembly - Modified

RELEASED DOCUMENT

ALTERED ITEM DRAWING

RI/RD88-256

103

QTY REQD	FIND NO.	PART OR IDENTIFYING NO.	FSCM	NOMENCLATURE OR DESCRIPTION	MATERIAL	SPECIFICATION
1	2	-5		EXTENSION	Ni 200 WIRE	ASTM
1	1	-3	85311	SPARK PLUG		MAKE FROM 6LA4907EE (5)
-1	—	7R035388				

PARTS LIST

NAS-9-25108

D. PONN

R. DEMINICO

Rockwell International Corporation
Rocketdyne Division

SPARK PLUG ASSY-MODIFIED

SIZE D | FSCM NO. 02602 | DRAWING NO. 7R035388

SCALE 2/1 | SHEET 1 OF 1

DO NOT SCALE PRINT

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY (*Leave blank*)

2. REPORT DATE
November 1988

3. REPORT TYPE AND DATES COVERED
Final Contractor Report

4. TITLE AND SUBTITLE
Space Station Hydrogen/Oxygen Thruster Technology

5. FUNDING NUMBERS
WU–481–01–02–00
NAS3–25142

6. AUTHOR(S)
G.L. Briley and R.S. Iacabucci

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
Rockwell International
Rocketdyne Division
Canoga Park, California 91303

8. PERFORMING ORGANIZATION REPORT NUMBER
E–13086

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Washington, DC 20546–0001

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
NASA CR–182280
RI/RD88–256

11. SUPPLEMENTARY NOTES
Project Managers, Phillip Meng and David Byers.

12a. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified - Unlimited
Subject Category: 20
Available electronically at http://gltrs.grc.nasa.gov/GLTRS
This publication is available from the NASA Center for AeroSpace Information, 301–621–0390.

12b. DISTRIBUTION CODE

13. ABSTRACT ***(Maximum 200 words)***

This report covers the effort expended by the Rocketdyne Division of Rockwell International in fulfilling the requirements of the Space Station Freedom Hydrogen/Oxygen Thruster Technology Program. The report includes the basis and the rationale for the design of the thruster, injector, and nozzle, discusses the test results, and presents the lessons learned, together with conclusions and recommendations for the development of the Space Station Freedom thrusters.

14. SUBJECT TERMS
Space Station Freedom; Hydrogen/oxygen engines; Injectors; Rocket nozzles

15. NUMBER OF PAGES
91

16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	

NSN 7540-01-280-5500

Standard Form 298 (Rev. 2-89)
Prescribed by ANSI Std. Z39-18
298-102

ASAP Record Display

Go To Body

Search | Next Term | Bottom of Page | Help | Logout

Additional help: Search Results | Viewing Documents | STI Help Desk

View more information about record by selecting, **Full ASAP Record**

Record 1 out of 1

Space station hydrogen/oxygen thruster technology

Final Report

19890017534 Nov 1, 1988 Technical Report 111p

Unclassified Unrestricted - Publicly Available

Availability: Issuing Activity;

Report No.: NASA-CR-182280; NAS 1.26:182280; RI/RD88-256

Contract No.: NAS3-25142

Task/Rtop No.: RTOP 481-01-02

Abstract: The effort expended by the Rocketdyne Division of Rockwell International in fulfilling the requirements of the Space Station Freedom Hydrogen/Oxygen Thruster Technology program is discussed. Included are the basis and the rationale for the design of the thruster, injector, and nozzle; the test results; and the lessons learned, together with conclusions and recommendations for the development of the Space Station Freedom thrusters.

Org Source: Rockwell International Corp.
Canoga Park, CA, United States
Rocketdyne Div.

Author/Editor: Briley, G. L.; Iacabucci, R. S.

Doc Language: English

Full ASAP Record Listing

Field	Data
Document ID	19890017534
Accession Number (1963-1997)	89N26905
Title	Space station hydrogen/oxygen thruster technology
Type of Progress Report	Final Report
Author/Editor	Briley, G. L.; Iacabucci, R. S.
Author Affiliation	Rockwell International Corp.; Rockwell International Corp.;
Author Affil.	

Location	Canoga Park, CA, United States; Canoga Park, CA, United States;
Publication Country	United States
Numeric Publication Date	19881101
Textual Publication Date	Nov 1, 1988
Document Language	English
Number of Pages	111p
Document Type	Technical Report
Report Number	NASA-CR-182280; NAS 1.26:182280; RI/RD88-256
Subject Category	SPACECRAFT PROPULSION AND POWER
Abstract	The effort expended by the Rocketdyne Division of Rockwell International in fulfilling the requirements of the Space Station Freedom Hydrogen/Oxygen Thruster Technology program is discussed. Included are the basis and the rationale for the design of the thruster, injector, and nozzle; the test results; and the lessons learned, together with conclusions and recommendations for the development of the Space Station Freedom thrusters.
Abstract Source	CASI
Major Terms - NASA Thesaurus	HYDROGEN; NOZZLE DESIGN; OXYGEN; ROCKET ENGINES; ROCKET NOZZLES; SPACE STATION FREEDOM; SPACE STATIONS; SPACECRAFT PROPULSION
Minor Terms - NASA Thesaurus	DESIGN ANALYSIS; INJECTORS; PERFORMANCE TESTS; THRUST
Contract No.	NAS3-25142
Project/Task No.	RTOP 481-01-02
Financial Sponsor	NASA;
Financial Sponsor Location	United States;
Financial Sponsor Type	NASA
Organization Source	Rockwell International Corp.;
Org. Source Sub-unit	Rocketdyne Div.;
Org. Source Location	Canoga Park, CA, United States;
Security Classification	Unclassified
Access/Distribution Restriction	Unrestricted - Publicly Available
Available Source	Issuing Activity;
Subject Category Code	20
Date Modified in ASAP	20010511
Date Loaded to ASAP	20010511

Search | Prev. Term | Top of Page | Help | Logout

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY (*Leave blank*)	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
	November 1988	Final Contractor Report

4. TITLE AND SUBTITLE

Space Station Hydrogen/Oxygen Thruster Technology

5. FUNDING NUMBERS

WU–481–01–02–00
NAS3–25142

6. AUTHOR(S)

G.L. Briley and R.S. Iacabucci

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

Rockwell International
Rocketdyne Division
Canoga Park, California 91303

8. PERFORMING ORGANIZATION REPORT NUMBER

E–13086

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

National Aeronautics and Space Administration
Washington, DC 20546–0001

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

NASA CR–182280
RI/RD88–256

11. SUPPLEMENTARY NOTES

Project Managers, Phillip Meng and David Byers.

12a. DISTRIBUTION/AVAILABILITY STATEMENT

Unclassified - Unlimited
Subject Category: 20

Available electronically at http://gltrs.grc.nasa.gov/GLTRS

This publication is available from the NASA Center for AeroSpace Information, 301–621–0390.

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

This report covers the effort expended by the Rocketdyne Division of Rockwell International in fulfilling the requirements of the Space Station Freedom Hydrogen/Oxygen Thruster Technology Program. The report includes the basis and the rationale for the design of the thruster, injector, and nozzle, discusses the test results, and presents the lessons learned, together with conclusions and recommendations for the development of the Space Station Freedom thrusters.

14. SUBJECT TERMS

Space Station Freedom; Hydrogen/oxygen engines; Injectors; Rocket nozzles

15. NUMBER OF PAGES

91

16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	

NSN 7540-01-280-5500

Standard Form 298 (Rev. 2-89)
Prescribed by ANSI Std. Z39-18
298-102